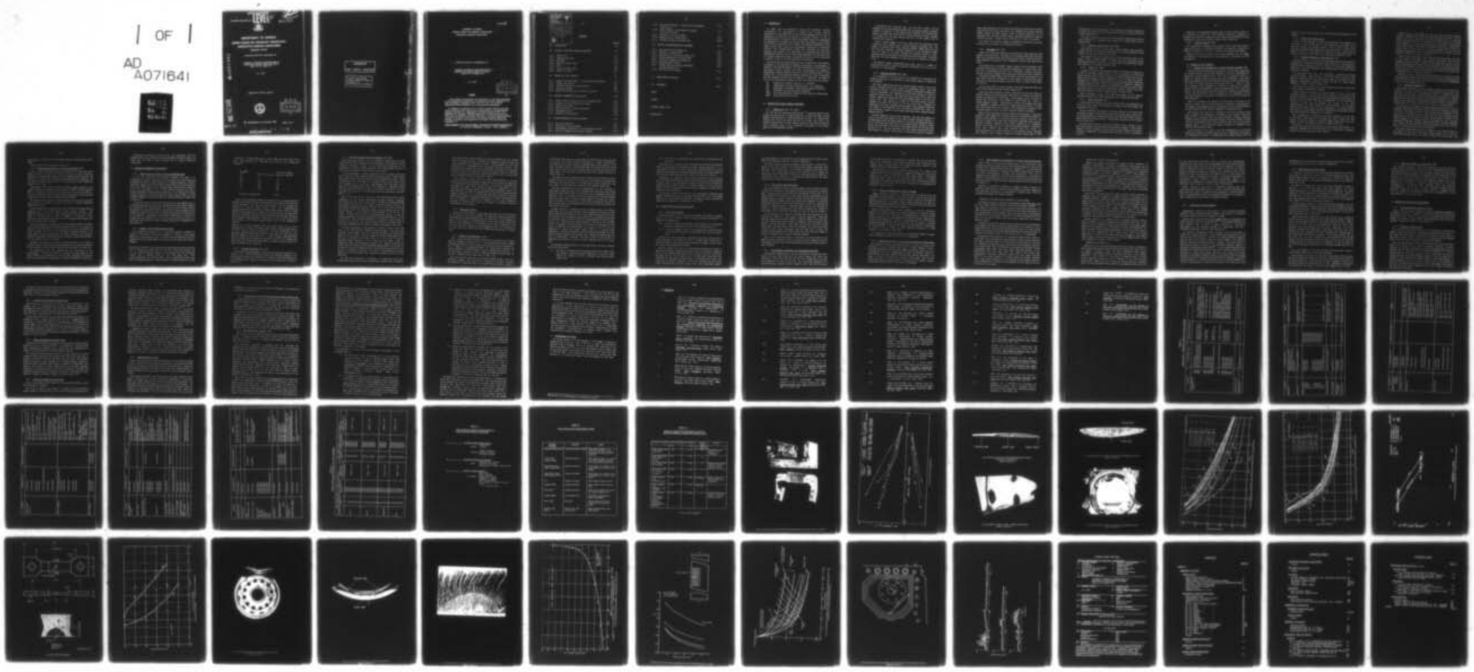
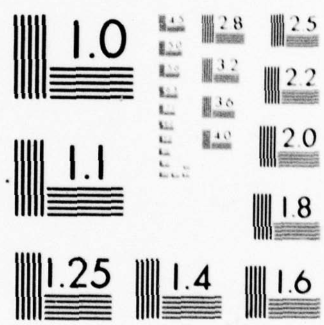
AD-A071 641 AERONAUTICAL RESEARCH LABS MELBOURNE (AUSTRALIA) F/G 1/3
A REVIEW OF AUSTRALIAN INVESTIGATIONS ON AERONAUTICAL FATIGUE D--ETC(U)
APR 79 G S JOST
UNCLASSIFIED ARL/STRUC-TM-303 NL

1 OF 1
AD
A071641



END
DATE
FILMED
8-79
DDC



MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

ARL-STRUC-TECH-MEMO-303

DEPARTMENT OF DEFENCE

DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION

AERONAUTICAL RESEARCH LABORATORIES

MELBOURNE, VICTORIA

Structures Technical Memorandum 303

A REVIEW OF AUSTRALIAN INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1977 TO MARCH 1979

G.S. JOST

Approved For Public Release.

APRIL, 1979

AR-001-356 610

DEPARTMENT OF DEFENCE

DEFENCE SCIENCE AND TECHNOLOGY ORGANISATION

AERONAUTICAL RESEARCH LABORATORIES

(14) ARL/STRUC-TM-303

(9) Structures Technical Memorandum 303

(6) A REVIEW OF AUSTRALIAN INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1977 TO MARCH 1979

(11) Apr 79

(12) 68p.

(10) G.S. JOST

SUMMARY

This document was prepared for presentation to the 16th Conference of the International Committee on Aeronautical Fatigue scheduled to be held at Brussels, Belgium on May 14 and 15, 1979. It is being distributed within Australia as an ARL Technical Memorandum.

A summary is presented of the aircraft fatigue research and associated activities which form part of the programs of the Aeronautical Research Laboratories, Commonwealth Aircraft Corporation Pty. Ltd., Department of Transport (Airworthiness Branch), Royal Australian Air Force and the Government Aircraft Factories. The major topics discussed include the fatigue of both civil and military aircraft structures, fatigue of materials and components and fatigue life monitoring and assessment.

POSTAL ADDRESS: Chief Superintendent, Aeronautical Research Laboratories, P.O. Box 4331, MELBOURNE, Victoria 3001, AUSTRALIA.

008 650 JOB

Accession For
NTIS GRA&I ☒
DDC TAB ☐
Unannounced ☐
Justification

By
Distribution/
Availability Codes
Dist. | Avail and/or special
A

CONTENTS

9.1 INTRODUCTION

In common with the situation in many other countries, economic circumstances and requirements in Australia are forcing critical reappraisal of many aspects of aircraft fleet management with a view to maximising the effective utilisation of existing aircraft. So far as airframe fatigue is concerned, these requirements demand the reassessment and sometimes the extension of the safe operating fatigue lives of given aircraft by all possible practical means. The necessary components of such means include load, strain and environmental monitoring on individual aircraft, more representative fatigue test evaluation of structure, component and material, the development of specialised NDI and crack refurbishment methods, and of course continuing effort in the elusive area of reliable predictive fatigue. Continuing activity in all of these areas is evident in many of the contributions to the Review.

This Review has been made possible by the co-operation of the author's colleagues at the Aeronautical Research Laboratories, the Department of Transport (Airworthiness Branch), the Royal Australian Air Force and in the Australian aircraft industry and their contributions are gratefully acknowledged. Unless otherwise stated the topics discussed refer to work carried out at the Aeronautical Research Laboratories; other information has been supplied officially by the Organisation indicated. The names of the various organisations referred to have been abbreviated as follows:

ARL	Aeronautical Research Laboratories, Melbourne
CAC	Commonwealth Aircraft Corporation Pty. Ltd. Melbourne
DOT	Department of Transport (Airworthiness Branch) Melbourne
GAF	Government Aircraft Factories, Melbourne
NZAIL	New Zealand Aerospace Industries Ltd, Hamilton, New Zealand
RAAF	Royal Australian Air Force

9.2 FATIGUE OF MILITARY AIRCRAFT STRUCTURES

9.2.1 Mirage IIIO (ARL, CAC, RAAF)

The 1975 and 1977 Australian ICAF Reviews [1,2] contained details of the fatigue test carried out on a Mirage wing under a flight-by-flight loading sequence representative of RAAF operational service. The demonstrated fatigue life of the test wing was such as to clear that item to the currently required life of type. However, since the test article comprised only a starboard wing, no test information was gained on the fatigue quality of the remaining airframe.

Calculations have indicated that the next most fatigue critical component is the main fuselage frame (No.26) carrying the wings in bending. Clearance of frame 26 to the required life of type for the aircraft must, however, be based upon the results of fatigue testing an actual airframe. It is hoped that the Swiss Mirage test will, in due course, help in answering this question.

Initial fatigue damage in both the Swiss and the Australian tests occurred as skin cracking from a fairing fastener attachment hole. The design of a suitable boron fibre reinforced plastic (BFRP) patch for this area has been completed. Although inspection of the Australian Mirage fleet for signs of the above cracking has revealed nothing to date, cracking has been discovered in the nearby fuel drain hole area. A suitable BFRP patch has been designed for this area and is detailed in Section 9.5.8.

The Mirage fatigue investigation has given rise to a number of associated activities. These are reported in Sections 9.4.5, 9.5.2, 3,6,7, 8 and 9.6.1 of this Review.

9.2.2 Aermacchi MB.326H (ARL, RAAF)

As foreshadowed in the last Review, the possibility of restoring the fatigue properties of cracked Macchi centre section booms by reaming has been examined.

Two booms, removed from service aircraft because cracks in one of the bracket attachment holes had reached the permissible depth, were made available by the RAAF. The holes were enlarged by progressive reaming from 5 mm to 8 mm diameter, thus removing the cracks and a minimum of 0.25 mm of apparently sound material. The booms were then fatigue tested under the tension- compression randomised block load sequence used in the full scale fatigue test. They endured an average of 540 blocks before failure, compared with about 500 blocks in the full scale test. Although the comparison is not exact (bending loads, present in the complete structure, were absent in the component test) the results indicate that, under the test conditions, reaming of the cracked holes effectively removed fatigue damaged material.

A further component test has been completed. This test was designed to relate the block load sequence so far used in Macchi fatigue testing to a corresponding flight-by-flight sequence. The latter was generated from a set of continuous time histories of n_z obtained during a series of flight trials, and conformed to the overall load spectrum for the aircraft (see Section 9.6.4.). The test specimen, which had already experienced about 1600 flights in service, failed after an additional 3060 test flights.

Using range-mean-pair cycle counting and fatigue data which had correctly predicted the full scale test result, the predicted life under flight-by flight loading was 4600 flights. This is sensibly identical to the component life. The failure mode, however, was quite different to those of previous failures. (Compare Fig.9.1 with Fig 9.1 of the 1977 Review).

As a result of this work, reaming of redeemable cracked centre sections is being put into effect in the RAAF Macchi fleet.

9.2.3 GAF Nomad (GAF, ARL)

Nomad is an Australian designed and manufactured twin turbo-prop STOL general utility aircraft. Two models are available; model N22B and model N24A of all up-weights 3855 kg and 4263 kg respectively. These aircraft are being employed in military, civil, geophysical, medical and surveillance roles.

The fatigue test on the Nomad [3] began in August 1976; the planned 45,000 simulated flying hours involving 45000 landings was attained by May, 1978. The test is continuing with the aim of achieving a primary structural failure and has now logged 72000 flying hours without any major incidents [4]. Similarity between models N22B and N24A allow for test results to be read across with due allowance for AUW, service conditions and minor structural variations.The applied loads are based on a mean cruise weight of 4173 kg.

Initial test results to date indicate that model N22B and N24A safe lives exceed 25000 and 16000 flights respectively.

Cracks have been located by visual inspection and critical areas are checked regularly using radiography, eddy current, ultrasonic and acoustic emission non-destructive techniques. Strain gauges in the vicinity of cracks have indicated no real change in stress levels as a result of fatigue damage or repair. In particular, cracks in the primary stub wing rib suggested a possible flexibility change which may result in stress level variations within the structure. No variations were observed, even following repairs to the rib, which restored original stiffness.

Repairs to the stub wing called for removal of the top skin. During this exercise, visual inspection of internal structure confirmed the absence of fatigue cracks adjacent to critical areas and internal structure. The majority of repairs have used the conventional aluminium alloy patches. Wherever possible, aluminium alloy repair schemes are of elementary design, and include the use of readily available mechanical fasteners. Boron fibre repairs have been employed when the cracks are non-representative or where difficulty is encountered using aluminium alloy repairs. Such a repair, placed on the wing leading edge at 5000

flight hours, is still serviceable. All aluminium alloy repair schemes are designed with the requirement of meeting the limited facilities available in more remote areas.

The test is continuing at the rate of approximately 800 flying hours per week; designers are confident that a safe life of 30000 hours will be demonstrated.

9.2.4 NZAIL Air Trainer CT4A (ARL, RAAF)

The CT4A Air Trainer is a two place, piston engined, fully aerobatic trainer aircraft of 1070 kg all-up-weight and is being produced in New Zealand for the Royal Australian Air Force.

The proposed full scale fatigue test on this aircraft is still in the future. Only a limited number of flight tests and associated strain measurements have been carried out, but have indicated strain levels higher than expected in the lower wing spar joint.

9.2.5. F111C (CAC, RAAF)

A Service Life Monitoring Programme (SLMP) for the F111C is being set up by CAC for the RAAF. Damage rates are to be calculated on a continuing basis using the output from multi-channel recorders fitted to two aircraft. These recorders sample flight parameters at up to 30 times per second. Stresses at 25 different control points are calculated using regression analysis based on flight trials data. Assignment of damage to components in aircraft without multi-channel recorders will be based on g-meter counts and pilot reports of aircraft configuration (weight, stores, fuel used, wing sweep angle) and type of flying (speed, altitude, duty).

Final development of the F111C SLMP and handover of the complete operating system to the RAAF is expected to be achieved during 1980.

9.2.6. CA.27 Avon Sabre (CAC)

Former RAAF Avon Sabres have been in service with the Indonesian Air Force since 1970. Some of the fleet are now close to the safe fatigue life originally calculated.

CAC has reappraised the fatigue life of each aircraft in the fleet using accumulated g-meter counts where available and average spectra where not. The fatigue analysis is complicated by the fact that some fatigue critical parts have been reinforced after a considerable fraction of the original life had been used.

The wing centre section is the most critical area for fatigue, and in some instances the calculated safe life has been consumed. A crack growth analysis has been carried out on this section, and an inspection technique developed. Inspection is required every 400 hours on the life-expired aircraft.

Inspection is by radiography because there are five layers of sheet to be inspected and access for other possible inspection techniques is limited. The aircraft is supported on the outer store hard-points during radiography to open any cracks and improve their visibility.

9.2.7. Canberra B.Mk.20 (RAAF)

The accummulation of damage in the fatigue critical lower centre section lugs of the Canberra is presently monitored by fatigue meter counts and an appropriate formula. No account is taken of any landing load damage.

Advice has now been received from British Aerospace that the damage attributable to such loads may not be significant. The matter is being followed up.

9.3 FATIGUE IN CIVIL AIRCRAFT

9.3.1 Fatigue life promulgation - General Aviation aircraft (DOT)

In January 1970 the Australian airworthiness certification requirements for General Aviation aircraft were amended to include a requirement for structural fatigue substantiation. Since then, retirement lives for a number of General Aviation aircraft have been promulgated. Lives which have been assessed in Australia and which differed from life limitations in the country of origin, have been reported previously.[1,2] Table 9.1 lists all retirement lives promulgated or assessed in Australia for General Aviation aircraft (excluding helicopters and gyrocopters) over this 9 year period. The table includes lives which have been assessed or re-evaluated in the 2 year period under review. An example of a life assessment carried out where adequate stressing data for the aircraft type did not exist has been completed.[5]

Where the retirement lives shown in the table differ from those of the country of origin, this is usually because of operational considerations. For example, the atmospheric gust environment in most of inland Australia is more severe than that shown by commonly used published data.

Past practice has been not to promulgate lives which exceeded 15,000 hours. However, the combination of generally good weather conditions prevailing in Australia, and the large distances between population centres, leads to relatively high utilisation rates and a number of light aircraft are achieving total times in service that were not previously thought likely. It has therefore become necessary to promulgate lives which exceed the above figure.

Also included in the table is the life promulgated by the Gliding Federation of Australia under delegation from the Department of Transport, for the Blanik L.13 glider. The Department has a growing concern over the

fatigue of world lead time metal and fibreglass gliders operated in this country.

9.3.2 Flight loads spectra (DOT)

Activity in the program of flight loads measurement has remained at a low level in the two year period. Some progress has been made in the currently topical and perhaps unusual area of glider flight loads; preliminary Fatigue Meter measurements from two Blanik L. 13 aircraft are shown in Figure 9.2. Also shown for comparison are the results from two glider tug aircraft some of which were previously reported in Ref.2.

9.3.3 Propeller fatigue and inspectability (DOT)

Included in Ref. 1 was a report covering fatigue problems which had been experienced in Australia with Harzell Y Shank propellers. Two instances of circumferential cracking in the fillet radius of the retention flange were described, as was the development and implementation of an eddy current inspection technique.

The manufacturer has since then introduced a manufacturing change involving pressure (cold) rolling of the fillet radius in lieu of shot peening. Operators have also been made more aware of the effects of, for example, overspeeding, vibration induced by wear of engine damper components, ground or object strikes, etc.

The cold rolling, and the eddy current inspections appear at this stage to have largely contained this problem.

Fatigue cracks initiating from stone damage pits are an old and recurring problem. In one instance which occurred in the period under review the propeller tip separated and the 160 mm portion penetrated the pressure cabin, slightly injuring two passengers.[6] It also severed an electrical wiring loom and deprived the aircraft of all radio communications. The pilot landed the aircraft safely, if in silence; however the incident could have been catastrophic. The fatigue failure initiated from a stone impact mark which had been detected but had not been completely removed by dressing with an abrasive.

Fig 9.3 shows numerous indentations and paint chipping on the rear face of the blade. Four locations, including the failure origin, had been dressed, in some of these the original damage indentations are still clearly visible. The fracture surface is shown in Fig. 9.4.

Efforts have been made to make operators more aware of this problem. The subjective impression is that these efforts are reflected by a gradually reducing rate of occurrence of these failures.

Another type of less common but none the less recurring propeller problem is examplified by the mid-span failure shown in Fig 9.5[7]. In this instance the failure initiated at a point of corrosion attack 0.75 mm long and 0.15 mm deep on the front or camber surface of the blade. The corrosion attack appeared to follow a path along a strain line within a single grain at the blade surface. This form of corrosion, although very serious as a stress raiser, is very difficult to detect visually. The rate of crack propagation appeared to be very rapid, possibly a few flights, thus the problem is difficult to detect "in esse".

Strain lines were evident in sets of parallel planes within the grains of both blades, indicating that plastic deformation had occurred subsequent to solution heat treatment. Both blades were reported to have been re-straightened following a previous accident, and this may account for the above evidence of cold work. There is also the likelihood that high residual tensile stresses will remain at the front face of the blade after re-straightening in a rearwards direction.

Presuming that the re-straightening operation was carried out within the manufacturer's prescribed limits, there is a case for recommending that these limits be revised, even to the extent of prohibiting re-straightening in this critical mid-span section.

9.3.4 Helicopter fatigue (DOT)

Helicopters, although in terms of numbers a minority on the Australian Register of Aircraft, continue to attract a disproportionate amount of time and effort in the field of airworthiness control. A number of fatigue problems have come under scrutiny, many of which originated in countries other than Australia and are therefore outside the coverage of this report. However, one significant failure which occurred recently in Australia is outlined here as it will no doubt be of interest to ICAF delegates.

The helicopter was engaged in off-shore operations between oil drilling rigs when sudden severe in-flight vibration dictated an emergency landing on the sea. Following recovery of the helicopter and disassembly of the rotor head, it was found that the main rotor yoke was completely broken through on one side and that the failure was by fatigue. Because of the lack of damage to the fracture surfaces and because the only load path redundancy is through a flexible tension strap which can take no bending, it is likely that the in-flight failure was partial and that the final failure occurred during the recovery operation.

The failure originated from one of a number of small stress raisers in the surface of the steel. The stress raisers appear to be a form of preferential pitting attack, the nature of which is not yet clear. The

investigation is still in its early stages and no conclusions can yet be drawn.

The fracture surface is shown in Fig. 9.6.

9.3.5 Manufacturing defects in aircraft structures (DOT)

Early in the period under review, an aeroplane on the Papua New Guinea Register became the first of its type to reach the initial inspection threshold for radiographic inspection of the wingmain spar lower cap. The aircraft is a modern twin turboprop pressurised executive/commuter, and the inspection was required as part of the overall fail safe inspection program for the type.

Predictably, the inspection found no fatigue damage; however it did reveal a manufacturing defect in the form of a "double- drilled" fastener hole in a critical area of the built-up spar cap. The critical nature of the spar in this area results in part from the termination of four titanium straps which reinforce three aluminium alloy elements.

On the basis that this was an isolated defect, the particular aircraft was allowed to continue in service, but with a reduced repetitive inspection interval.

Several months later, another (Australian) aircraft of the same type was inspected and it too was found to have similar defects - one "double-drilled" hole and one which completely missed the pilot hole. Since the problem could no longer be regarded as isolated, a fleet inspection was arranged; of the four aircraft in Australia/New Guinea, all four carried this type of defect.

It is not possible to determine from the radiographs whether the defect is in the aluminium spar, the titanium straps, or both. If the defect is confined to the titanium it is not so serious as it is near the end of the strap. However, if it is in the continuous aluminium elements it is particularly serious; delegates will no doubt be aware of an in-flight fatigue failure which resulted some years ago from a double-drilled hole, in a non-critical spar cap location.

Probably the only way to resolve this question will be to remove a fastener from one of the affected holes; this has not yet been done because of the complexity of the structure and the lack of access to the spar cap.

The manufacturer has been advised and has contributed to the investigation. A particular batch of aircraft has been identified by serial number, and inspection of these has been promulgated by Service Bulletin, specifying the radiographic technique developed in Australia.

Proposals for corrective action have yet to be formulated. This will probably differ with individual aircraft depending on defects present, and could take the form of repair schemes and/ or special repetitive inspections.

9.4 FATIGUE OF COMPONENTS AND MATERIALS

9.4.1 Batch-to-batch variation in 2L.65 aluminium alloy

In the 1973 Australian ICAF Review[8] reference was made to an investigation in which 13 batches of 2L.65 aluminium alloy were being used to examine the batch-to-batch variability in fatigue properties. Rotating cantilever fatigue tests on both unnotched specimens and notched specimens of $K_t = 3.65$ (involving a total of 1300 specimens) have now been completed. It is clear from Figs 9.7 and 9.8 that different batches can exhibit markedly different fatigue strength properties. At 10^8 cycles the average fatigue strengths for unnotched specimens ranged from 120 MPa to 170 MPa (17000 to 25000 psi); and for notched specimens from 60 MPa to 70 MPa (8500 to 10000 psi).

A disturbing feature shown by this investigation is the apparent lack of correlation between the various batch parameters which have been compared. No correlation was found between the unnotched and notched fatigue strengths of the various batches; nor between the fatigue strengths and the fatigue notch sensitivities, static strength properties or microstructures of the material. Furthermore, considerable differences in scatter in fatigue life were exhibited by the various batches and this could not be correlated with scatter in static strength properties or differences in microstructure.

9.4.2 Fatigue of thick aluminium alloy lugs

The investigation reported in ref. 9 has been extended to examine the effects of hole surface finish and frequency of cycling on the fatigue behaviour of thick pin/lug joints. As before the test material was 2L.65 aluminium alloy, the lugs being 31.75 mm thick and the steel pins 19 mm in diameter.

Hole surface finish: For this investigation the holes were bored to provide both a coarse hole finish (CLA 27 m) and a fine hole finish (CLA 1.9 m). Fatigue tests were made at a cyclic frequency of 1 Hz in repeated tension with a minimum stress of 23.4 MPa (3400 psi). Both constant-amplitude and six-load range lo-hi-lo program load tests were made at maximum stresses of 51 MPa (7400 psi); 67 MPa (9700 psi); 105 MPa

(15200 psi); 137 MPa (19900 psi); 165 MPa (23900 psi) and 195 MPa (28300 psi). Two severities of "spectrum" were used in the program-load sequences:

S_{max} (MPa)	Cycles per program	
	"Severe" spectrum	"Moderate" spectrum
195	1	1
165	112	28
137	248	62
105	314	138
67	304	320
51	70	500
Cycles per program	1049	1049

Under constant-amplitude conditions the mean life of coarse-finish hole specimens was found to be significantly greater than those of the fine-hole finish specimens at the top four stress levels, but not significantly different at the two lowest stress levels. The "severe"spectrum program-load tests also indicated a significantly greater life for coarse finish hole specimens, but no significant differences in lives were found under the "moderate" spectrum. This behaviour under spectrum loading is not altogether surprising when the differences in the ratios of the cycles corresponding to the various maximum stress levels of the programs are compared.

Frequency of cycling: Fatigue tests have been carried out at cyclic frequencies of 1, 4 and 16 Hz under similar fatigue loading conditions to the hole surface finish investigation. The only difference in the specimens compared with those referred to above was that the holes were machine reamed to a fine finish of 2.1 m CLA. Constant- amplitude tests have not indicated any significant "frequency" effects, but program-load tests under the "severe" spectrum suggest a slight increase in mean life with increasing cyclic frequency.

9.4.3. Bolted joint tests (DOT, CAC)

The fatigue testing programme on bolted joint specimens referred to in Section 9.5.2. of the 1977 Australian Review has now been completed. The results show fatigue lives consistently in excess of published design data for aluminium joints. Details of the tests and the results are in course of publication.

9.4.4 Water-displacing corrosion inhibitors (DOT,ARL)

Water-displacing organic corrosion inhibitor preparations are being used by both aircraft manufacturers and operators in an attempt to control in-service corrosion. Because of the lubricating properties of the preparations, concern has been expressed as to their possible effects on the fatigue performance of bolted and riveted joints.

In one investigation, constant-amplitude ($R=+0.1$) fatigue tests have been carried out under repeated tension on several types of 8-bolt bolted joints made from 3.7 mm thick 2024-T3 Alclad aluminium alloy sheet. These included both "low" and "high" load transfer joints, using high and low bolt clamping forces in each case. Complementary tests were made on each type of joint assembled with either "dry" components or components coated with the corrosion inhibitor. Most of the tests were made at a cyclic frequency of 2.5 Hz but, at low alternating stresses, additional tests were made at 17 Hz.

For fully-torqued bolted joints the water-displacing fluids proved detrimental to the fatigue performance at high constant- amplitude alternating stresses (lives of about 0.04×10^6 cycles) presumably because they reduced the ability of the joints to transfer load by friction. At intermediate alternating stresses (lives of 0.1×10^6 cycles) the fluids were found to have no significant effect on the fatigue lives of the joints. However, at low stresses (at lives between 2×10^6 and 20×10^6 cycles) a reverse effect to that at high stresses was apparent in that the fatigue performance of joints coated with the fluids was superior to that of dry joints. This has been attributed to a reduction in the severity of fretting in the joints. For loosely torqued joints, the fatigue lives of wet joints exceeded those of similar dry joints by factors of up to four.

A second investigation is in progress on smaller 12-bolt joints made from 3.2 mm thick DTD 683 aluminium alloy extrusion. Only high load transfer, fully clamped joints are involved and these are being tested under $R = +0.1$ conditions. A cyclic frequency of 40 Hz is being used so that data can be obtained at very long lives, i.e. 80 to 100×10^6 cycles. Although similar behaviour to that found with the 8-bolt joints is exhibited at high and intermediate stresses, it is apparent that the dry joints exhibit a "fatigue limit" phenomenon whereas this is not evident in the case of corrosion- inhibited joints. No failures have occured in "dry" joints at alternating stresses of less than 41.5 MPa (6000 psi) whereas failures in "wet" specimens have occurred at stresses as low as 31 MPa (4500 psi).

The Australian Department of Transport, in common with some other airworthiness authorities, has in the past been concerned over the possible

long term effects of water displacing corrosion inhibitors on the fatigue life of structural joints. A fatigue test program on simple riveted lap joint specimens was reported in Ref.1. This program has now been completed; of 72 specimens tested 58 produced valid test results. (The remainder were excluded because of inadvertent overload, test equipment malfunction etc.) The final test results are shown in Table 9.2 and Fig. 9.9, and as previously reported confirm the results of Ref.10. Additional tests on bolted joints are reported in Ref.11.

The validity of these tests, and their applicability to aircraft structures, has beeu the subject of some discussion between the Department of Transport and the Australian airlines. It has quite reasonably been contended that these "new" specimens tested in a laboratory, do not represent "old" aircraft joints in an operational environment. There is no doubt that this contention is supported by airline service experience with commercial products, i.e. it appears that the benefits gained in corrosion prevention generally outweigh any degradation in fatigue characteristics. This is difficult to quantify and attempts to do so could perhaps be regarded as "looking a gift horse in the mouth". No further testing is contemplated at this stage.

9.4.5 Superposed notches

Inspection of blind holes in a wing spar boom indicated that the bottoms of the holes were incorrectly formed. The spherical radius specified on the drawing had a small superposed notch at the centre. To investigate the effect of this defect on fatigue behaviour, axial load fatigue tests were made on 2L.65 aluminium alloy specimens of the type illustrated in Fig 9.10. The results are shown in the S/N diagrams of Fig. 9.11 from which it is clear that such a defect can cause significant reductions in fatigue life.

9.4.6 Fatigue of landing wheels (DOT)

Ref. 2 outlines fatigue problems experienced in Australia with landing wheels on General Aviation aircraft. In the period under review, fatigue problems with wheels on large transport aircraft have attracted attention. As an example, one investigation (Ref. 12) covered three separate but similar failures of Goodyear wheels fitted to Douglas DC-9 aircraft. In each case about a quarter of the tyre retaining flange had broken away from the hub, a typical failure being shown in Fig. 9.12.

The three fracture surfaces, one of which is shown in Fig. 9.13, were remarkably similar. One fracture had initiated at pitting corrosion approximately 0.15 mm deep, Fig. 9.14, and when final failure occurred at

11183 flights had grown to a circumferential length (around the bead seat) of 140 mm. The second failure originated at a small discontinuity, considered to be an oxide inclusion, approximately 0.5 mm long and 0.15 mm deep. The wheel failed at 10700 flights when the fatigue crack was 120 mm long. There was no evidence of a stress-raiser at the fatigue origin of the third wheel, this crack was 170 mm long when final failure occurred at 10622 flights.

Each wheel had been inspected using a fluorescent dye penetrant method between 69 and 137 flights prior to failure. Crack growth determined from one failure and shown in Fig. 9.15, showed that a crack at least 60 mm long was present in each wheel when inspected, and remained undetected.

The wheel manufacturer's manual recommendation is for ultrasonic or eddy current inspection, and it is considered most likely that these cracks would have been detected prior to failure if the recommended inspections had been in use. It is apparent that the fluorescent dye penetrant method cannot be relied upon to detect fatigue damage in the bead seat radius. This is because of the particular stress and environmental situation in this location and in no way suggests that the method lacks effectiveness at other wheel locations.

Fatigue failures of transport aircraft landing wheels has become a significant cost and safety problem. Aircraft wheels are basically "safe life" items which are treated in service on a "safety-by-inspection" basis. The majority of aircraft wheels used in Australia are designed and produced in the United State of America to Technical Standard Order TSO C-26b issued by the U.S. Federal Aviation Administration. Basically this standard calls for the qualification of wheels by a roll test to 1,000 miles at rated load. In addition, the wheels have to be shown by test to have the capability of making 100 normal energy stops on a suitable wheel dynanometer. Some aircraft manufacturers are now requiring wheel and brake manufacturers to design their products for a higher performance than the current airworthiness standards. The FAA is also in the process of upgrading the standards to include at least a 2000 mile roll test at rated load.

In assessing fatigue behaviour of aircraft wheels two basic problems have been encountered:-

(i) the lack of a methodology for estimating the life of a wheel under the variable loading conditions experienced in service, using results of a single load level or several load levels rolling fatigue test.

(ii) the lack of a methodology for statistically assessing service failures.

Regarding (i), a literature search has failed to disclose any meaningful work on the analytical assessment of wheel fatigue generally. It is considered that this is a topic needing some research effort to promote better understanding of the fatigue problem of light alloy wheels which are also finding increasing application outside the aircraft industry.

Regarding (ii), incomplete failure data, consisting of time to failure on failed units and differing times on unfailed units, are called multiply censored. Data on aircraft wheels operating in the field for example are always multiply censored because of frequent changes of tyres and other practical considerations which demand many spare wheels in constant use in normal airline operations.

The analysis of wheel failures has been meaningfully assessed using Nelson's hazard plotting for incomplete failure (Ref. 13). This method offers useful application possibilities in a number of areas requiring the quantitative assessment of incomplete failure data.

9.5 STUDIES RELATING TO FATIGUE CRACKING

9.5.1 Crack tip behaviour

The physical model of the load/COD relationship introduced in Section 9.4.5 of the 1977 Australian Review has been further developed.[14] It comprises two components:

(i) an elastic component, dependent on the stiffness of the unyielded material surrounding the plastic crack tip material and

(ii) a plastic component based on the stress/strain characteristics of the uncracked material representing the crack-tip plastic zone.

A series of crack propagation experiments have been carried out on copper specimens to investigate the effect of added plastic constraint in low cycle fatigue. The constraint was provided by a number of steel rods, in a fixture surrounding the specimen.

Previous results from tests on unconstrained specimens had shown an excellent correlation between crack growth rate and the plastic component of the crack opening displacement (COD). Results from the constrained specimens did not lie on these curves. The hysteresis loops were then re-plotted by subracting the added elastic constraining force and it was found that the plastic component of the strain obtained from these wider loops correlated well with the previous data. These experiments suggest

that the parameter controlling the crack propagation rate is plastic work, which is unchanged by addition of elastic constraint.

The study is presently being extended in two directions. In the first, a study of low cycle fatigue, in titanium alloy engine compressor disc materials has been initiated. It is intended to monitor plastic work cycle by cycle. The second study concerns the validation of a scheme for simulating elastic constraint in which the loads that would be applied to the purely elastic members of a structure are subtracted in the servo controller by voltage analogue methods.

9.5.2 NDI research and development

An active program of research and development in NDI is being continued at ARL. On the research side emphasis is being concentrated on acoustic NDI. A theoretical study of scattering from defects is under way - it is hoped that this will eventually lead to more reliable quantisation of the measured defect. Acoustic emission (AE) has been used for crack location during laboratory fatigue tests and airborne AE equipment is currently undergoing evaluation. AE is also being used as a tool for studying the effects of microstructure in aluminium alloys, and for the detection of corrosion. Image enhancement has been used to simplify the reading of radiographs and is also being developed for study of fracture surfaces.

On the practical side, special techniques have been developed for the detection of defects in a wide range of aircraft components. These have included light beam (aircraft canopies), magnetic rubber (centre sections and main landing gear shock struts), ultrasonics (centre sections, wing planks, internal corrosion in MLG lower drag struts), and eddy current (wing skins, aircraft wheels, skins of helicopter tail pylons and brake calipers); eddy current techniques have also been used to detect and monitor cracks beneath boron fibre reinforced plastic patches bonded to aircraft components and to detect disbonded regions.

Work is also being undertaken to determine the effectiveness of the magnetic rubber technique in detecting cracks in steel components on which non-magnetic surface layers have been applied (e.g. metal plated layers or paint schemes).

9.5.3 Fractographic examination of Swiss Mirage fractures

Fractographic examinations of fracture surfaces of the main spar and a section of wing tank skin from the Swiss fatigue test have been completed.[15] The results indicate that the crack propagation in the main spar commenced very early in the test. The crack growth appeared to be relatively smooth, having no abrupt changes in rate. The fracture surface

of the skin material was less amenable to analysis than the spar fracture due to poor definition of the surface features. The indications are, however, that the crack growth rate changed in a manner similar to that observed during the ARL Mirage fatigue test.

The solutions for the stress intensity factors for the cracks in the components have not yet been obtained. Comparison between the crack growth rates in the spars of the ARL and Swiss tests shows a significant difference when related to the applied flight blocks. A possible contribution to the observed difference could lie in the different stress intensities at the growing crack tips due to the different crack geometries produced in the two tests.

9.5.4 Fracture toughness and residual strength

Damage tolerance concepts are currently being introduced into the development and maintenance of military aircraft. New structure is now assumed to contain flaws and structural safety thereafter protected by safety-by-inspection procedures. These changes emphasise the need for determining a range of material parameters, fatigue crack growth rates and fracture toughness being the more important.

Factors influencing the fracture toughness of aircraft structural materials and their influence on residual strength are presently being studied. It has been found that the maximum stress intensity during fatigue cracking, K_f, on the subsequently measured fracture toughness K_Q, is particularly important. Results for aluminium alloy 2014 show that K_Q remains unchanging for K_f values up to about 0.7 K_Q; K_Q, however, increases for K_f values above 0.7 K_Q. In the typical case of a fatigue crack growing to critical size, the higher K_Q values observed correspond to a 50% increase in the critical crack size.

It is proposed to use these results in interpreting residual strength tests performed on fatigue cracked aircraft components.

9.5.5 Load history effects - fatigue crack retardation in D6AC steel

The influence of single and multiple peak overloads on fatigue crack retardation in high strength D6AC steel is being examined further. Increased retardation occurs with the number of overload cycles, N_o, up to a maximum of 15 to 20. An approximately linear relationship appears to exist between the stress intensity overload ratio and the logarithm of the number of cycles of retarded (or delayed) crack growth. For overload ratios exceeding 2.3, crack growth at the lower level is, however, delayed indefinitely ($N_o > 10^6$ cycles). The work is continuing.

9.5.6 Fibre composite reinforcement of cracked aircraft structures

Work at ARL on the practical application of high performance fibre composites, such as carbon or boron fibre reinforced plastics, has centred on their use to modify defective metallic components.[16] In this approach, the composite is adhesively bonded to the component selectively in regions of strain concentration. Although most applications have been to fatigue-cracked components, the scope of the procedure is, as indicated in Table 9.3, much greater than this. A list of some current applications and projects is given in Table 9.4 which includes aspects listed in areas A and C of Table 9.3.

Research is currently in progress in a number of areas to support the practical exploitation of selective reinforcement of defective components, with particular reference to the patching of fatigue cracks. This work includes both design and material aspects.

Analytical and experimental procedures in patch design

Two design procedures for patch reinforcement have been employed; these are the Integral Equation method[17] and the Finite Element method[18]. In both cases the reduction in stress intensity produced by patching, the shear stress distribution in the adhesive and the stress distribution in the reinforcement can be obtained. Fig.9.16 shows typical relationships obtained for stress-intensity reduction by the Finite Element method and Fig. 9.17 the type of design curves that can be obtained by the Integral Equation method.

Although the analytical procedure is much faster and uses considerably less computer time than the Finite Element method, it is limited to fairly simple situations, such as centre-notched panels. Most of our patch design studies are based on the Finite Element method which has been developed to represent the adhesive layer and the crack tip in 2 or 3 dimensions.

Transmission photo-elasticity of transparent epoxy plastic models is being used to give experimental confirmation of predicted stress intensity for repair situations where the reinforcement is just ahead of the crack tip. Attempts have also been made to obtain the stress-intensity reduction in fatigue-cracked 2024, T3 aluminium alloy panels by measuring fatigue crack-growth rates before and after repair with BFRP patches. So far difficulties have been found with this method, since under most representative patching conditions (within the safe design envelope Fig.9.17) the stress intensity is reduced below the fatigue threshold level. Under these conditions only the minimum stress-intensity reduction can be estimated as no further crack propagation occurs. However, work is continuing on this aspect.

Thermal and residual stress studies

Despite the numerous advantages of advanced fibre composites for selective reinforcement of metals, one important disadvantage must be considered in any application. This is the problem of thermally induced stress which results from the low expansion coefficient of the composites compared to metals ($5 \times 10^{-6}\,{}^{o}C^{-1}$, BFRP and $0.1 \times 10^{-6}\,{}^{o}C^{-1}$ CFRP compared to $23 \times 10^{-6}\,{}^{o}C^{-1}$ for aluminium alloys). The severity of the residual stress after patching depends on a number of factors which include the adhesive bonding temperature, the thickness and geometry of the composite and metal substrate, and the degree of constraint to thermal expansion offered by the surrounding cool structure during patching. In general, due to constraint, the level of residual stress will be very much less than the maximum theoretically possible.

Early work[19] had shown that the tensile residual stress introduced in aluminium alloy 7075 components by BFRP patching did not produce stress corrosion problems; indeed, the procedure has been successfully used to repair stress corrosion cracks, Table 9.4. Studies on the effect of the residual stress caused by composite reinforcement on the fatigue of plain 7075-T6 specimens showed that a slight reduction in fatigue strength occurred; estimation of this reduction on the basis of the Goodman construction was found to be conservative. Recent work has been undertaken to measure the residual stress distribution at the crack tip in CFRP reinforced centre-notched 7075-T6 aluminium alloy specimens.

Thermal fatigue of the adhesive layer is another potential problem. Even in subsonic aircraft, temperature fluctuations from $-50^{o}C$ to over $+100^{o}C$ can occur, resulting in shear stress cycling of the adhesive layer. Studies of the significance of this effect are under investigation[20] for BFRP reinforced 7075-T6 aluminium specimens. So far the results indicate that, for the configuration used, cycling from $+120^{o}C$ (the bonding temperature) to $-50^{o}C$ for over 1000 cycles causes no damage in the adhesive layer; however fatigue cracking does occur in the adhesive when a lower cycle temperature of about $-100^{o}C$ is used.

Adhesive bond durability

Adhesives for the crack-patching application were chosen on the basis of their fatigue and stress-relaxation properties together with their ability to produce acceptable values of these properties when simple, non-chemical, pre-bonding surface treatments were used on the adherend surfaces[21]. Tests resulted in the choice of adhesive AF126 (an epoxy nitrile adhesive produced by the 3M company) for all crack patching applications. Recent work has centered on chemical surface treatments which could provide improved bond durability with adhesive AF126, yet still

be used under field conditions. Tests to evaluate the environmental durability have been based on the Boeing Wedge Test, which has been shown to give results which correlate well with service behaviour of adhesive bonds in aluminium aircraft structures. Table 9.5 sets out the various surface-treatments evaluated on aluminium alloy 2024-T3 and the results obtained in terms of crack growth in standard times. It can be seen that the local anodise in phosphoric acid gives a considerable improvement on non-chemical surface treatments. Further improvements appear to be obtained by a combination of alumina grit blasting and phosphoric acid anodising; this treatment is currently under investigation.

Unlike aluminium alloys where no durability problems have been encountered with bonded BFRP repairs in service, durability problems have been found with repairs to magnesium alloy components. Currently similar tests to those described above for aluminium are underway in an attempt to find an improved field procedure for magnesium alloys.

9.5.7 Structural fracture mechanics

Considerable attention is being paid to stress analysis problems associated with cracked structures. Broadly, this involves a combination of finite element and fracture mechanics concepts.

Earlier work on the use of the finite element method for determining the stress intensity factor in cracked sheets[22,23] is now being extended to the more difficult case of cracks in solid sections.[24] A theoretical investigation of the residual strength of a pin-loaded lug with a through-the-thickness crack has been carried out.[25]

One problem of substantial current interest is that of the design of repairs for cracked structures. It has already been demonstrated in ARL that the use of an overlay, or "patch", of a high performance composite material such as boron fibre reinforced plastic (BFRP), is an effective means of repairing a cracked structure. In support of the work outlined in the previous section, a study of the structural mechanics aspects of patching a cracked sheet has been made[26]. A finite element method has been developed for predicting both the reduction in stress intensity factor achieved by a given patch design and, also, the values of the shear stresses in the adhesive bonding the patch to the sheet; these are the main structural factors in determining the performance of the patch. The method has been applied to practical situations and is now being extended to a study of the patching of cracked solid sections. The results of these

investigations are particularly encouraging and indicate that the repair of cracks in thick sections is a distinct possibility.

9.5.8 BFRP patches for Mirage wings

As reported in Section 9.2.1, cracks have been discovered in the drain hole regions of the lower surfaces of Australian Mirage aircraft. One possible refurbishment option is to make use of boron fibre reinforced plastic (BFRP) patches. The design and development of such a patch has been undertaken at ARL.

The general configuration of the patch being developed is shown in Fig.9.18 . It is a uni-directional seven layer laminate, the fibres running perpendicular to the direction of crack propagation. The patch contains all holes necessary for attachment of the drain hole cover, but the drain hole in the patch itself has been much reduced in size; it is a slot approximately 12mm x 3mm. The patch is to be covered by aluminium foil, bonded to provide environmental protection.

A finite element analysis of a cracked and patched wing panel, with the crack extending 50mm on the spar side of the drain hole and 30mm on the root rib side was undertaken. This indicated that the patch reduced the stress intensity factors at the tips of the 50mm and 30mm cracks to 13% and 2% of their unpatched values, respectively.

Constant amplitude fatigue tests have been carried out on cracked and patched 2024T3 Alclad aluminium alloy sheet specimens 889mm x 508mm x 3.6mm. The specimens had the full size drainhole and the full set of twenty four rivet and bolt holes for attaching the drain hole cover; some specimens also had simulated spar bolt holes. Typically, the initial cracks in the sheet were 30mm long on either side of the drain hole. Some difficulties were encountered with early specimens due to shortcomings in the original patch application procedure. This has been modified and, in the latest test, the patch prevented any crack growth for over 400,000 cycles in the stress range 11.7 MPa to 95.8 MPa. In this test the crack was contaminated with a fuel/water mixture throughout; in the last stages of the test the specimen was subjected to an occasional maximum stress of 158 MPa.

It is now planned to carry out spectrum load tests under simulated environmental conditions on additional specimens.

The procedure to be used in the field for the patching of fleet aircraft has been successfully tested on a trial wing. The first actual repairs of two cracked wings on a fleet aircraft are now in progress. This will be followed by repair of a second aircraft in 1979. Assuming no difficulties are encountered, the repair of the fleet will commence later in 1979.

9.5.9 Residual strength of cracked lugs. (DOT)

The residual strength of cracked lugs continues to be one of the most difficult unsolved problems of analytical structural mechanics and one of considerable practical concern to the industry. Contact with the leading aircraft manufacturers indicates that some work is still being done on this problem on a low priority basis, primarily because of the cost of the finite element analyses which are now generally accepted as the best approach to the problem. Considerable difficulties are also being encountered in correlating these analyses, which are basically two-dimensional, to test data. The use of Bueckner's weight function approach to calculate the stress intensity factor in a variable stress field is reported by the McDonnell Douglas Aircraft Company[27] to yield encouraging but not precise results.

9.6 FATIGUE LIFE MONITORING AND ASSESSMENT

9.6.1 Life monitoring for Mirage fin (CAC)

As part of the continuing fatigue monitoring and assessment programme on the Mirage airframe, it is planned to establish a service life monitoring system for the fin. This will be carried out in parallel with the Aircraft Fatigue Data Analysis System (AFDAS) being developed by ARL and British Aerospace (Australia) reported in Section 9.6.6.

9.6.2 Analysis of censored data

For some years ARL has been interested in the analysis of censored data samples. These arise naturally in fatigue tests where there are runouts or more than one mode of failure. Typically, failures at a given site are precluded by prior failures elsewhere, and in general both the observed failures and the runouts constitute censored data samples.

There are several methods of analysing such data but that used at ARL is based on the solution of the maximum likelihood equations. Iterative solutions for the mean and variance of censored log normal data were first obtained over 20 years ago,[28] the iterations depending on a function arising naturally from the censorship ratio. The procedure has now been computerised and the program package extended to have the capability of treating several samples with a common variance - this allows analysis of censored analogues of all the classic linear models on the analysis of variance. The same programs now produce values of minimised likelihoods so that different models may be compared by ratio tests. Weibull distributed data may also be analysed.

The manner in which extremely sparse data are dealt with by the programs is presently under investigation, in particular the bias of the estimators. This seems to be small for 'reasonable' samples but increases markedly when nearly all of the data comprise runouts.

9.6.3 Variability in structural fatigue life

Some years ago a survey was carried out on the variability in the fatigue lives of aluminium alloy aircraft structures tested under programmed and random loading[29]. Since that time some additional data have been published and these have been included in a more recent reappraisal of the total data, comprising over 200 individual test results.

The data have been treated as log normal and as Weibull distributed. As originally found, fighter spectra give rise to a comparatively low variability in fatigue life whilst for transport spectra the variability is rather higher. The standard deviations of log life are 0.09 and 0.14 respectively, whereas for the same data treated as Weibull distributed, the corresponding dispersion parameters are 5.9 and 3.9. The latter figure is close to that taken by the Boeing Company (dispersion parameter=4) as being typical of civil aircraft.

9.6.4 Flight sequencing for fatigue tests

A method has been developed to produce a representative flight by-flight sequence for use in fatigue testing[30]. It is appropriate to the situation where comparatively extensive fatigue meter data are complemented by limited, but representative continuous flight histories.

The annotated fatigue meter data sheets are first used to set up event frequency tables describing the frequency with which each given type of flying is followed by each of the others, and by itself. The measured flight histories are then arranged into a flight-by-flight sequence of specified length (typically 200 flights) by drawing at random from these tables. The process is largely computerised, and includes automatic adjustment for rounding errors and count corrections resulting from compressing perhaps years of fatigue meter data into a repeating 200 flight sequence. Examples of typical flights are shown in Fig. 9.19.

9.6.5 Structural fatigue and reliability

Since the last Review which reported on work detailed in Ref.31, there have been several advances.

The theory is based on the division of fatigue into two successive stages, damage or crack nucleation, then a period of cracking terminated by

fracture. The last stage, fracture, has previously been viewed as a problem in reliability theory alone, to be computed from prior results of structural fatigue. The risk of fracture has now been incorporated into the main theory. This has produced simplifications and suggested a new measure of fatigue sensitivity. The improved approach also includes other risks besides fatigue, thus allowing the above measure of fatigue sensitivity. It also allows for a realistic inclusion of inspections.

The work above has gone furthest for single-crack models[32]. For several critical locations interacting through crack growth, consideration of mappings from critical damage to initial lives to crack length vectors, using a corrected product theorem, has produced a complete solution but without allowance for inspections. This multi-crack model postulates a finite number of possible cracks which do not join. The next extensions envisaged will remove these restrictions. It is expected that the extensions will involve elements of mathematical programming.

A FORTRAN 10 program is being developed for the life distribution and inspection behaviour of single-crack structures. One of the test models for this deals with previous reliability and inspection estimates for the Macchi training aircraft. Because these models ultimately relate basic results on initiation and crack growth rate to the fatigue of assembled structures, the agreement with experiment depends critically upon basic input data. Current work at ARL, and elsewhere, on crack growth retardation is important here, as is a clear-cut definition of initial failure. However, there is a particular lack of initial life or cumulative damage data for use in contributory Neuber-Topper-Wetzel analyses.

9.6.6 Range-mean-pain counter

Recent literature indicates that range-mean pair (or rainflow) counting of load or strain cycles is gaining wide acceptance as the preferred method of cycle counting for fatigue life estimation and monitoring. An Aircraft Fatigue Data Analysis System (AFDAS), which uses this counting method, is being developed by British Aerospace (Australia) from an original concept by ARL.

This equipment processes on-line data from up to eight channels (strain gauges, accelerometers or other electrical transducers), by detecting peaks and troughs, quantised at 16 levels, pairing them according to the range-mean algorithm, and summing the count into run-volatile memory. The information is read in computer-compatible format onto a cassette tape, the frequency of interrogation being set by the needs of the user. When fully developed the instrument will allow economic monitoring of any airframe

component. It is now in the final stages of development and is expected to be available during 1979.

9.6.7 Fatigue prediction using standardised loading sequence data

An investigation is continuing on the feasibility of using fatigue data generated under realistic loading sequences to replace conventional S-N data in fatigue life prediction. At least some of the difficulty associated with accurate fatigue life prediction is certainly associated with what are loosely called 'load history' effects.

Conventional S-N data totally suppress load history information and allowance for it must therefore be made in the prediction model. The approach investigated here uses data in which load history effects are automatically included (for example flight-by-flight loading data) in predicting fatigue performance for like or similar types of sequences. Such an approach is practical only given an adequate fatigue data bank derived from representative and typical loading sequence tests.

9.6.8 The "95/95 Life" Concept (DOT)

This concept, as put forward by Boeing in Ref.33 and described briefly in Ref. 1, attempts to quantify the worth of service experience relative to full-scale fatigue testing, in demonstrating adequate fatigue life in fail-safe aircraft.

The Australian Department of Transport has been examining this approach, and has published a report[34] covering the theoretical derivation and the underlying assumptions. (Boeing in Ref.33 do not derive the method theoretically, but give what appears to be empirically-based procedure for calculation). The concept is developed to a point where it is acceptable, with certain reservations, for use as a monitor of inspection intervals of fail-safe aircraft types. Graphs presented in the report clearly demonstrate the quantitative way in which service experience becomes relatively more and more important as the fleet ages, compared with the fatigue test. Fatigue testing is still necessary to cover the early service (of the order 1/3 to 1/2 the life), and for another reason, discussed shortly. The Department has made a trial application of the method to the Boeing 747 fleet.

The 95/95 life concept quantifies the satisfactory service experience to date. It is applicable to those structural details where no cracks are known to have initiated so far. It estimates the "life" as a function of total satisfactory experience, (in hours or landings), over the fleet. This "life" carries a 95% probability of survival of the detail from new, with a scale parameter for the underlying assumed Weibull distribution of times to failure, estimated at 95% confidence. The use of the concept as an

inspection monitor includes Boeing's "Lead the Fleet" concept, which means that provided the 95/95 life exceeds achieved flight time since new, the inspection schedule is satisfactory. (This of course assumes that inspection techniques are good enough to find cracks that are there. The Boeing 707 tailplane fatigue failure at Lusaka in 1977 shows the need for conservatism in this area). The "life" estimate grows with increasing service experience, provided the detail remains defect-free. This does not reflect any feeling that the structure improves with age - such a concept would of course have to be rejected. However it is recognised that the modern designer of large transport aircraft is successful with some of his design - perhaps even with most of it! (But never would even he claim success with all of it!). Thus it can be expected that some parts of the structure will not deteriorate, and details which have not so far deteriorated are members of that population of parts - with 95% /95% probability/confidence - provided they have not flown beyond the 95/95 life estimate. In due course some (5%) details will go defective, and these will drop out of 95/95 consideration at that time. Either they will have a preventive repair - fleet modification - and can possibly go back onto a "95/95" program, zero times at modification; or they will become a "recurrent defect", found by inspection at regular intervals, whose length is fixed by consideration of propagation rates, detectability, etc. So as the aircraft gets older, some parts of the structure do in fact fail in spite of 95/95, and it is on these occasions we can be glad about the fail-safe properties of the design.

The Department's version of the concept differs from Boeing's in the following respects:

(a) Its input data is hours (or landings etc.) at last inspection of each aircraft, not current achieved flight hours. This implies use of the concept to obtain 95%/95% probability/ confidence in nil crack at the estimated life, instead of in nil failure; "failure" being equated with progression of a crack to "complete or obvious partial failure failure of a single principal structural element" (FAR 25.571, pre-amendment 25-45).

(b) It examines the level of safety involved. Since shown to be equivalent to a probability of fifty nine in sixty, the balance of 1 in 60 is clearly too high a probability for a crack which might progress rapidly to a catastrophic failure. It could be acceptable however for cracks which are for the moment benign - i.e. in a fail-safe structure. This leads to the question of what level of probability of catastrophic failure is achieved if 95/95 is used for a fail-safe structure.

(c) Looking at the possibility of progression of a crack beyond failure of a "single principal structural element" - which could be termed catastrophic, or nearly so - whatever the name for the condition, an acceptable level of 1 in 1000 is suggested. This is equivalent to 99.5%/99.5% probability/confidence. It is possible, with certain broad assumptions, to calculate the propagation time required for a crack to develop from a "safe" failure to the "catastrophic" level, if the latter is to carry 99.5%/99.5 probability/confidence implicitly whilst the former meets 95/95 explicitly. The answer turns out to be of the order of 1/3 of the mean time to crack initiation - a very long propagation time, which implies very good crack resistance, probably using multiply- redundant structure.

(d) Thus the Department's version requires protection against the catastrophic failure, at a suggested level of 99.5/99.5, in addition to 95/95 protection against the safe failure. It is thought that on present experience this condition can only be demonstrated as being met by use of full-scale fatigue test aircraft, subjected to continuation of its load-spectrum program after judicious cutting of single principal structural elements, (as was done for the Boeing 747).

(e) Boeing in explaining the procedure for calculation of the 95/95 life, state that "a failure has conservatively been assumed to have occurred". The Department's theoretical derivation makes no such assumption, yet independently obtained the same numerical answers as Boeing's method (for the same numerical input data). This work was done prior to reading the Boeing report. Boeing's procedure is just a procedure, and talk of assuming a failure is only a useful aide-memoire to the procedure, with no relevance to the actual assumptions needed for the derivation. This must be stated to avoid the misleading feeling of vast conservatism achieved by "assuming a failure even though we have not had one".

The Department applied the concept as explained above, to the Boeing 747 world fleet inspection history to September 1976. This was believed valid after examining the results of Boeing's load cycling of their fatigue test aircraft after sawcutting; the condition of sufficiently slow crack propagation, following a safe failure, was believed met, rendering a 99.5/99.5 life to catastrophic failure less critical than a 95/95 life to safe failure. At the time of this work there was a general clamour for Boeing to extend the 747 fatigue test to two lifetimes. The work showed

that on the most optimistic assumptions possible (e.g. with regard to the value of the shape parameter), 95/95 would be just met by the past inspections to date and proposed inspection program into the foreseeable future, with a two-lifetime test (assuming a 60,000 hour "life" or "service objective")

It is thus believed that the 95/95 concept as outlined has potential application to future generations of aircraft, and could assist in resolving questions of length of fatigue test vs probable fleet size and service life, for fail-safe aircraft types known to have sufficiently slow crack propagation times. The fail-safe provision permits us to live with the occurrence of cracks occasionally, and not, as is argued by manufacturers, to eliminate a fatigue test requirement (although it does in fact substantially reduce the required length of an adequate test). Whether 95/95 for a "safe failure", and 99.5/99.5 for a "catastrophic failure" provide truly adequate levels of safety, taking also into account the probability of non-detection of existing cracks, requires close scrutiny by the aviation community as a whole.

9.7 BIBLIOGRAPHY ON FATIGUE

The Second Volume of the Bibliography on the Fatigue of Materials, Components and Structures - 1951 to 1960 by J.Y. Mann[35] was published by Pergamon Press in November 1978. * It contains 5903 references on the subject and brings the total in the first two Volumes to nearly 10000. Work is proceeding steadily on Volumes 3 and 4 which will cover the periods 1961 to 1965 and 1966 to 1970 respectively and each will contain over 5000 references. It is hoped that Volume 3 will be published by late 1980.

*Volume One of the Bibliography was published in 1970.[36]

9.8 REFERENCES

1. Jost, G.S. A review of Australian investigations on aircraft fatigue during the period April 1975 to March 1977. Minutes of the Fifteenth Conference of the International Committee on Aeronautical Fatigue. Published by Laboratorium fur Betriebsfestigkeit (edited by O. Buxbaum and J. Franz) during 1978.

2. Mann. J.Y. A Review of Australian investigations on aircraft fatigue during the period April 1973 to March 1975. Minutes of the Fourteenth Conference of the International Committee on Aeronautical Fatigue. Published by the Swiss Federal Aircraft Establishment (edited by J. Branger and F. Berger) during 1976.

3. Barker, M. Fatigue Test Specification, Government Aircr. Factories. Test Specification report no. N22-7221, Oct 1976.

4. Hutton, R.A. Preliminary Fatigue Test Report, Government Aircr.Factories, Nomad Project Note N2/68, 1979.

5. Esson, G.P. and Douglas, R.B. Airtourer wing static test and fatigue life evaluation. Dept. Transport, Airworthiness Branch, Aeronautical Engineering Report SM-86, Dec. 1977.

6. Metallurgical factual report on a failed propeller blade tip from Aero Commander 680 FL(P) aircraft VH-BTN. Dept. Transport, Specialist Report X-18/77, 1977.

7. Metallurgical factual report on a failed Sensenich propeller from Piper PA28 aircraft VH-PYL. Dept. Transport, Specialist Report X-12/78, 1978.

8. Mann, J.Y. A review of Australian investigations on aircraft fatigue during the period April 1971 to March 1973. Minutes of the Thirteenth Conference of the International Committee on Aeronautical Fatigue held at London, July 1973. Royal Aircr. Estab. Tech. Rep. 73184, March 1974.

9. Mann, J.Y.; Harris, F.G. and Revill, G.W. Constant-amplitude and program-load fatigue tests at low cyclic frequencies on thick aluminium alloy pin joints. Dept. Defence, Aeronaut. Res. Labs, ARL Structures Report 365, Sept. 1977.

10. O'Neill, P.J. and Smith, R.J. A short study on the effect of a penetrant oil on the fatigue life of a riveted joint. Royal Aircr. Estab. Tech. Report No. 73134, Feb. 1974.

11. Keays, R.H. Fatigue tests of bolted joints of 2024 aluminium alloy. Part 4 - Phase 3. Results and conclusions. Comm. Aircr.Corp. Report AA.381, Mar. 1979.

12. Metallurgical examination of three failed main landing wheel outer hubs from Douglas DC-9 aircraft. Dept. Transport, Specialist Report X-20/77, 1977.

13. Nelson, Wayne. Hazard plotting for incomplete failure data. J.Quality Contr^l, 1, (1), Jan. 1969.

14. Bullen, F.P., Retchford, J.A., Rogers, C.B. and Wicks, B.J. A model of crack-tip behaviour for fatigue life determination. Aircraft Structural Fatigue. Proceedings of a Symposium held in Melbourne, 19-20 Oct 1976. Dept. Defence, Aeronaut. Res. Labs., ARL Structures Report 363, ARL Materials Report 104, April, 1977.

15. Goldsmith, N.T. Fractographic examinations relevant to the F + W Mirage fatigue test. Dept. Defence, Aeronaut. Res. Labs., ARL Materials Tech. Memo. 371, Aug. 1978.

16. Baker, A.A. A summary of work on applications of advanced fibre composites at the Aeronautical Research Laboratories, Australia. Composites, 1978, 9 (1) pp 11-16.

17. Davis, M.J. Stress intensity reduction by strategic reinforcement. Proceedings of 1977 Conference of the Australian Fracture Group, 1977, pp 90-101.

18. Jones, R. and Callinan, R.J. Finite element analysis of patched cracks. J. Structural Mechanics, 1979, 8 (2).

19. Baker, A.A. and Hutchison, M.M. Fibre composite reinforcement of cracked aircraft structures. Proceedings 33rd Annual Technical Conference of the Plastics Industry, 1978, Section 17-E.

20. Baker, A.A. Hawkes, G.A. and Lumley, E.J. Fibre composite reinforcement of cracked aircraft structures - Thermal-stress and thermal-fatigue studies. Proceedings of the Second International Conference on Composite Materials, Metallurgical Society of AIME, Toronto, 1978.

21. Baker, A.A. Evaluation of adhesives for fibre composite reinforcement of fatigue-cracked aluminium alloys. Proceedings 10th National SAMPE Technical Conference, 1978, 10, pp 397-415.

22. Jones, R. and Callinan, R.J. A finite element method for calculating stress intensity factors in cracked sheets. Dept. Defence, Aeronautical Res.Labs.,, ARL Structures Report 360, Jan. 1976.

23. Jones, R. and Callinan, R.J. On the use of special crack tip elements in cracked elastic sheets. Int. J. of Fracture, v.13, pp. 51-64, 1977.

24. Jones,R and Callinan, R.J. A special crack tip element for three-dimensional crack problems, Dept. Defence, Aeronaut. Res. Labs., Structures Report 374, Nov. 1978.

25. Callinan, R.J. Residual strength of a cracked lug. Dept. Defence, Aeronaut. Res. Labs., ARL Structures Note 442, Oct. 1977.

26. Jones, R. and Callinan, R.J. On the stress analysis of patched cracks, Dept. Defence, Aeronaut. Res. Labs., Structures Report 367, Jan. 1978.

27. Impellizeri, L.F. and Rich, D.L. Spectrum fatigue crack growth in lugs. McDonnell Aircraft Company Report MCAIR 75-006, 1975.

28. Ford, D.G. Maximum likelihood estimators for truncated samples from normal populations. Dept. Supply, Aeronaut. Res. Labs., Report ARL/SM264, Oct. 1958.

29. Jost, G.S. and Verinder, F.E. A survey of fatigue life variability in aluminium alloy aircraft structures. Dept. Supply, Aeronaut. Res. Labs., Report ARL/SM 329, Feb. 1971.

30. Gratzer, L.R. Generation of a representative load sequence for the fatigue testing of Macchi MB. 326H spar booms. Dept. Defence, Aeronaut. Res. Labs., ARL Structures Note 450, Jan.1979.

31. Ford, D.G. The development of the theory of structural fatigue. Aircraft Structural Fatigue. Proceedings of a Symposium held in Melbourne, 19-20 Oct., 1976. Dept. Defence, Aeronaut. Res. Labs., ARL Structures Report 363, ARL Materials Report 104, April, 1977.

32. Ford, D.G. Reliability and structural fatigue in one-crack models. Dept. Defence, Aeronaut. Res. Labs., ARL Structures Report 369, July 1978.

33. Thompson, P.A. A procedure for estimating the demonstrated fatigue life of airplane structure from fleet service experience. The Boeing Company, Document No. D6-41246TN, 1973.

34. Young, P.C. "95/95" - A probabilistic method for monitoring structural inspection intervals. Dept. Transport, Aeronautical Engineering Report SM-95, July, 1978.

35. Mann, J.Y. Bibliography on the fatigue of materials, components and structures 1951 - 1960. Pergamon Press, London, 1978.

36. Mann, J.Y. Bibliography on the fatigue of materials, components and structures 1838 - 1950. Pergamon Press, London, 1970.

TABLE 9.1

FATIGUE LIVES - AUSTRALIAN GENERAL AVIATION AIRCRAFT

Aircraft		Retirement Life	Inspection Threshold	Remarks
Aero Commander Rockwell	112	11,200 hours		
	500B	12,100 hours/23,200 hours	8,400/14,000 hours	Life & inspection threshold depend on wing strap mod. status, certificated max. operating weight, & if low level operation is involved.
	500U	12,100 hours/23,200 hours	5,600/14,000 hours	
	500S	12,100 hours/23,200 hours	5,600/14,000 hours	
	560F	16,300 hours	15,000 hours	Low level survey operations.
	560F	24,100 hours	17,800 hours	
	680F	11,600 hours		Low level survey operations.
	680F	17,000 hours		
	680FLP	25,000 hours	20,000 hours	External strap modification.
	680FL	8,100 hours/10,400 hours	5,800/7,500 hours	Low level survey operation - External strap mod. & different max. operating weights.
	680FL	11,200 hours/14,400 hours	8,800/11,300 hours	External strap mod. & different max. operating weights.
	680T	27,100 hours	18,500 hours	External strap modification.
	680V	23,400 hours	17,000 hours	External strap modification.
	680W	23,400 hours	17,000 hours	External strap modification.
	681	23,400 hours	17,000 hours	External strap modification.
	690	12,900 hours		
	690A	12,900 hours		
Beagle B121 Pup		-	2,200 hours	Centre spar inspection.
Beagle B206		11,000 hours		

Aircraft	Retirement Life	Inspection Threshold	Remarks
Beech 18	10,000 hours or 10 years from modification; or 20,000 hours from original manufacture.		Mandatory strap modification.
Beech C23	100 hours of aerobatics		Interim conditional aerobatic life limitation.
Beech 65-A80	14,000 hours	4,600 hours	Outer wing attach fitting inspection.
65-B80	10,000 hours	8,700 hours	" " " " "
65-8200	13,200 hours	11,000 hours	" " " " "
65-80	-	4,900 hours	" " " " "
65	-	5,900 hours	" " " " "
70	13,200 hours	11,000 hours	" " " " "
65-A90	10,000 hours	7,600 hours	" " " " "
65-B90	10,000 hours	7,600 hours	" " " " "
65-C90	10,000 hours	7,600 hours	" " " " "
65-E90	13,600 hours	9,700 hours	" " " " "
A100	13,600 hours	11,400 hours	" " " " "
200	11,300 pressurised flights		Fuselage retirement.
Bellanca 8KCAB	1,100 hours of aerobatics		Wing front strut aerobatic life-actual aerobatic time.
BN-2 Islander	20,000 hours		
BN-2A Mk III Trislander	11,200 hours 13,600 hours		10,000 lb. gross weight aircraft. 9,300 lb. gross weight aircraft.
CA-28 Ceres	14,000 hours	3,000 hours	Mandatory strap modification.

Aircraft	Retirement Life	Inspection Threshold	Remarks
Cessna 310/320	14,900 hours		
401/402	10,000 hours		
421B/421C	8,600 hours		
Cropmaster YA-1	5,500 hours		3,530 lb. gross weight aircraft.
	6,500 hours		3,250 lb. gross weight aircraft.
DH104 Dove	3,400 hours/5,000 hours		Wing & spar boom depending on type of operation.
	14,000 hours		Wing & spar boom life after modification.
	17,500 hours		Dove series 1-6 & Riley Dove Series 1-6 wing & spar boom life after modification.
	1,800 hours/2,500 hours		Wing centre spar boom (lower).
	6,700 hours/10,000 hours		Wing centre spar boom (lower) after Mod. 538.
	40,000 hours		Wing centre spar boom (lower). after Mod. 779.
DHA-3 Drover	6,000 hours		Mk. 2 & 3 aircraft wings.
	8,000 hours		Mk. 2 & 3 aircraft wing with Mod. 60.
	8,000 hours/18,000 hours		Mk 2 & 3 aircraft wings with Mod. 119
	15,000 hours/21,000 hours		Mk 2 & 3 aircraft wings with Mod. 105.

Aircraft	Retirement Life	Inspection Threshold	Remarks
DHA-3 Drover (contd.)	5,500 hours		Mk. 2 & 3 aircraft wing centre boom (Al alloy)
	25,000 hours		Mk. 2 & 3 aircraft wing centre boom Mod (steel)
	4,850 hours		Mk. 3A & 3B aircraft wings.
	6,850 hours		Mk. 3A & 3B aircraft wings with Mod. 60.
	13,000 hours/19,000 hours		Mk. 3A & 3B aircraft wings with Mod. 105.
	4,500 hours		Mk. 3A & 3B aircraft wing centre boom (Al alloy).
	25,000 hours		Mk. 3A & 3B aircraft wing centre boom Mod. (steel).
DHC-1 Chipmunk	8,000 hours/9,000 hours		Wing root fitting (Al alloy)
	22,000 hours		Wing root fitting Mod (steel)
	10,000 hours		Wing centre section lower boom (Al alloy)
	30,000 hours		Wing centre section lower boom Mod (steel)
	15,000 hours		Wing root attachment links
	30,000 hours		Wing.
			Note: These lives are adjusted by Role Factors - (a) Normal usage 1.0 (b) Training usage 2.5 (c) Aerobatic usage 4.0
DHC-2 Beaver	20,000 hours	2,100 hours	Lift strut - early design.
	2,480 hours	2,100 hours	Lift strut - early design agricultural operations.

Aircraft	Retirement Life	Inspection Threshold	Remarks
DHC-2 Beaver (contd.)	25,000 hours		Lift strut - later design agricultural operations.
	30,000 hours		Strut attachment fittings in agricultural operations.
DHC-3 Otter	20,000 hours		Wing lift strut.
	2,500 hours/10,000 hours		Wing lift strut, low level operations.
DHC-6 Twin Otter Series 1, 100 & 200	25,000 hours		Wing.
	-	20,000 hours	Wing with Mod. 6/1117
	16,000 hours		Wing strut.
	30,000 hours		Wing strut with Mod. 6/1172
	25,000 hours		Fuselage frame 218.8.
	30,000 hours		Fuselage frame 218.8 with Mod. 6/1173.
Series 300	33,000 hours	15,000 hours	Wing with Mod. 6/1117..
	30,000 hours		Wing strut with Mod. 6/1172.
	30,000 hours		Fuselage frame 218.8 with Mod. 6/1173
Embraer EMB110	10,000 hours		Wing attachment fittings.
Fletcher FU-24	-	1,000 hours	Mandatory, strap modification.
Fuji FA-200	1,700 hours of aerobatics		Aerobatic life limitation - actual aerobatic time.
G.A.F. N22 Nomad	14,000 hours		Wing strut fitting.
Grumman American AA-5	12,000 hours		
Helio Courier	2,000 hours		Unreinforced wing centre section spar & attach fittings.
Meta Sokol L40	4,000 hours		

Aircraft	Retirement Life	Inspection Threshold	Remarks
Mitsubishi MU-2	8,200 flights		Outer wing spar joint fitting.
Morava L200	5,000 hours		
Piper			
PA18, PA20 & PA22	2,000 hours	500 hours	Wing lift strut attachment fitting.
PA18, PA20 & PA22	1,000 hours	500 hours	Wing lift strut attachment fitting for seaplanes.
PA23-250(6)	13,000 hours		
PA28-235	12,500 hours		
PA31-310	11,000 hours		
PA31-350	13,000 hours		
PA31P	14,000 hours		
PA31T	12,000 hours		
PA32	11,000 hours		
PA36	2,350 hours/4,500 hours		Life depends on mod. status.
SAAB Safir	5,000 hours		Wing attach bolts and fin and tailplane attach fittings.
Swearingen SA26	7,500 hours	1,500 hours	Wing spar inspection.
	-	3,000 flights	Fuselage inspection.
SA226	-	5,400 hours	Wing spar inspection.
	-	3,000 flights	Fuselage inspection.
Victa/A.E.S.L. Airtourer	12,000 hours		Aircraft undertaking competition aerobatics attract a factor of 20 on actual aerobatic time.
Blanik L-13 Glider	3,750 hours		Gliders launched predominantly by winch.
	4,000 hours		Gliders launched predominantly by aerotow.

TABLE 9.2

FATIGUE TEST RESULTS-RIVETED LAP JOINT SPECIMENS

	MAX. STRESS LEVEL = 131 MPa (19,000 psi)			MAX. STRESS LEVEL = 76 MPa (11,000 psi)		
Penetrant	Specimen No.	Life To Failure (Cycles)	Mean Life, Standard Deviation of Log Life	Specimen No.	Life To Failure (Cycles)	Mean Life, Standard Deviation of Log Life
	1	81670		11	318000	
	3	20805		33	340000	
	24	29700		34	436000	
LPS-3	25	25500	32870, 0.192	35	335700	338350, 0.120
	26	25800		36	337700	
	27	21600		37	200800	
	28	30500		50	475100	
	51	42100				
	52	48700				
	4	33610		13	402000	
	5	25630		29	538600	
	19	59300		30	257500	
WD-40	20	24000	29020, 0.138	31	166000	329510, 0.165
	21	29200		32	343000	
	22	20300		46	319340	
	23	30700		57	416000	
	59	30360				
	60	21570				
	61	34600		69	165800	
	62	34400		70	364800	
	63	23300		71	268300	259480, 0.143
Boeshield	65	28700	31230, 0.082	72	279200	
	66	29100				
	67	40100				
	6	62280		14	663000	
	8	45970		15	817000	
	38	75260		16	822000	620420, 0.097
	39	79100		42	603380	
DRY	40	52080	65660, 0.086	43	418430	
	41	77620		44	531930	
	55	71690		45	573200	
	56	70000		47	640400	

TABLE 9.3

APPLICATIONS OF SELECTIVE REINFORCEMENT TO MODIFY EXISTING METALLIC COMPONENTS

A	Stiffen Underdesigned Regions	
	reduce	(deflection (flutter
	increase	(static strength (buckling strength (fatigue strength
B	Restore Strength or Stiffness	
	after	(blending out corrosion (blending out flaws (expiration of nominal fatigue life
C	Reduce Stress Intensity	
	in regions	(with flaws (badly designed (badly manufactured (damaged in handling (with battle damage (with fatigue or stress corrosion cracks (current work)

TABLE 9.4

CRACK-PATCHING AND REINFORCEMENT STUDIES

AIRCRAFT COMPONENT	PROBLEMS	STATUS
Hercules Wing Plank	Stress-Corrosion cracks	BFRP repairs made to 11 aircraft now RAAF standard procedure (3½ years Service history)
Macchi Main Landing Wheel	Fatigue-cracking	BFRP repairs made to 4 wheels now RAAF standard procedure (2.5 years Service history)
Nomad Wing Skin (ARL Fatigue Test)	Fatigue-cracking	BFRP repair; no further crack propagation for 59000 flying hours
Nomad Door Frame (ARL Fatigue Test)	Fatigue-cracking	BFRP repair; no further crack propagation for 53000 flying hours
Fin	Acoustic fatigue	BFRP repairs recently made
Landing Wheel	Fatigue-cracking	BFRP repairs under development
Wing Panel	Fatigue-cracking	BFRP reinforcement developed for service application
Landing Wheel	Low fatigue life	BFRP/CFRP reinforcement under development
Gust Probe	Vibration	CFRP reinforced reduced vibration to acceptable level
Landing Gear Lever	Fatigue and overload failure	BFRP reinforcement under development

TABLE 9.5

RESULTS OF WEDGE TEST EXPERIMENTS ON 2024-T3 ALUMINIUM ADHERENDS, BONDED WITH ADHESIVE AF126

Surface Treatment	Crack Propagation (inches)		Fracture Mode	Comment
	1 hr	10 hr.	c = cohesive a = adhesive	
Hand abrade with 100# Al_2O_3	1.95	2.8	a	Standard field repair pre-treatment
Hand abrade with 100# Al_2O_3 + primer	2.53	3.15	a	
Grit blast with 50 µm Al_2O_3	1.82	2.52	a	Process previously used for crack patching repairs
Grit blast with 50 µm Al_2O_3 + primer	2.05	2.75	a	
Local anodise with phosphoric gel	1.37	1.45	partially c	Boeing suggested field repair pre-treatment
Local anodise with phosphoric gel + primer	1.35	1.39	partially c	
Chromic acid etch + phosphoric bath anodise (Boeing production specification BAC5535)	1.29	1.32	c	Factory production bonding treatment without primer *

* Not a field treatment

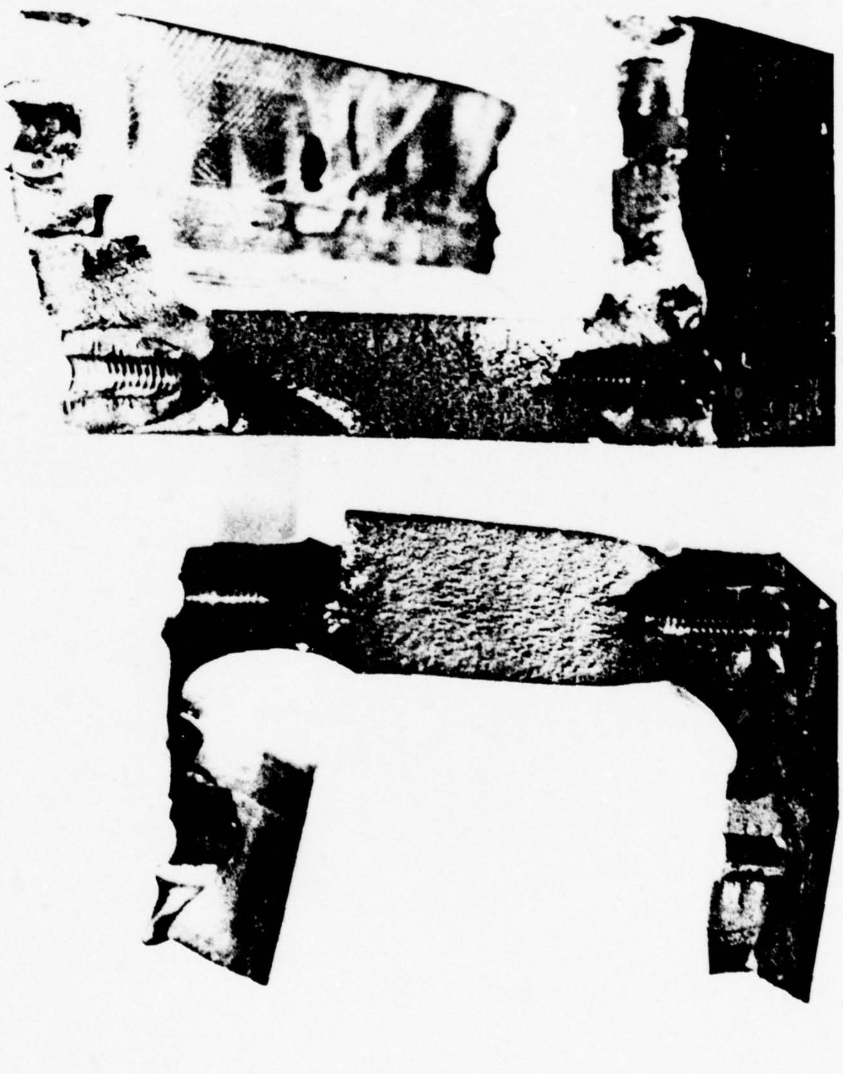


FIG. 9.1 FAILURE IN MACCHI CENTRE SECTION UNDER FLIGHT BY FLIGHT LOADING

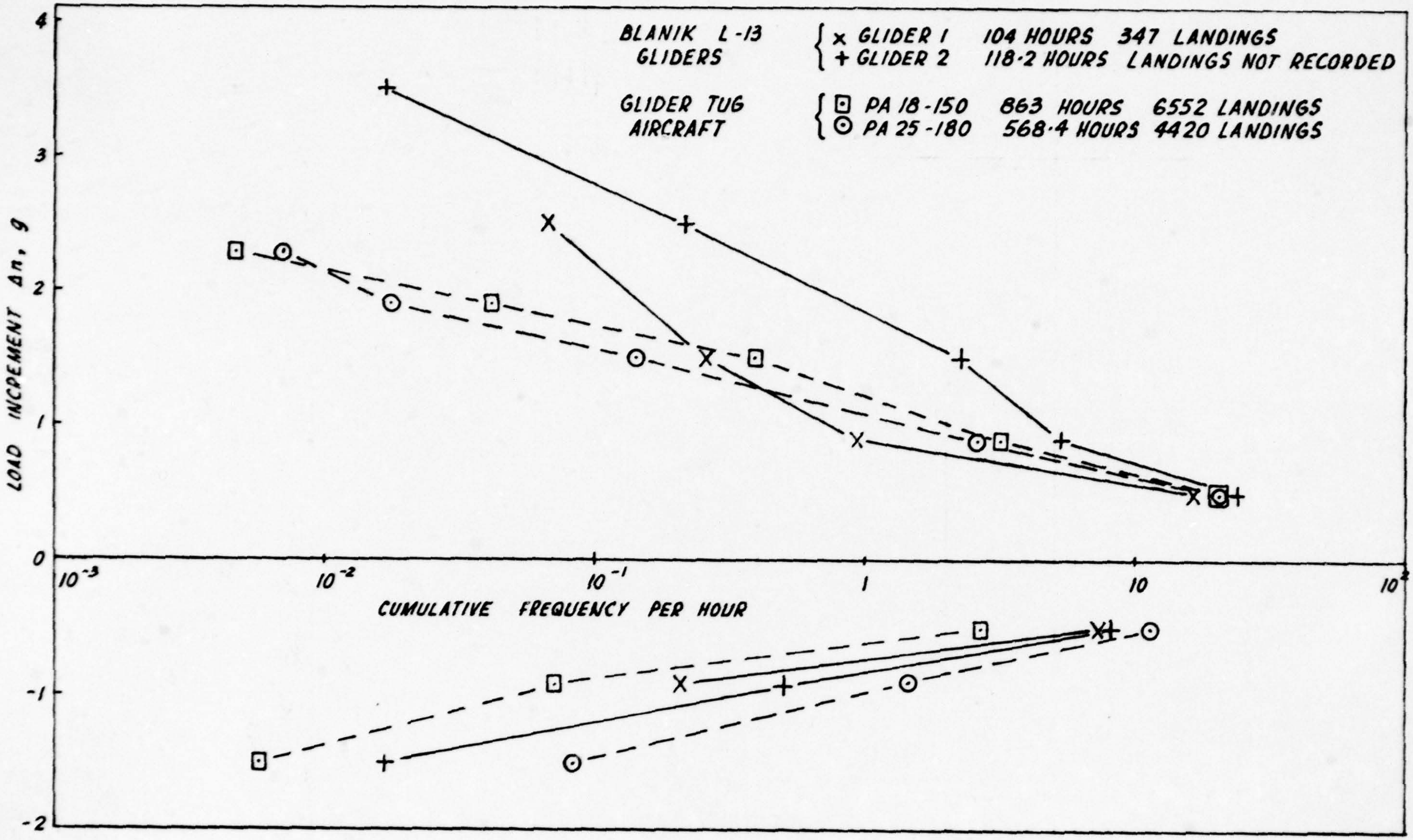


FIG. 9.2 FLIGHT LOADS SPECTRA

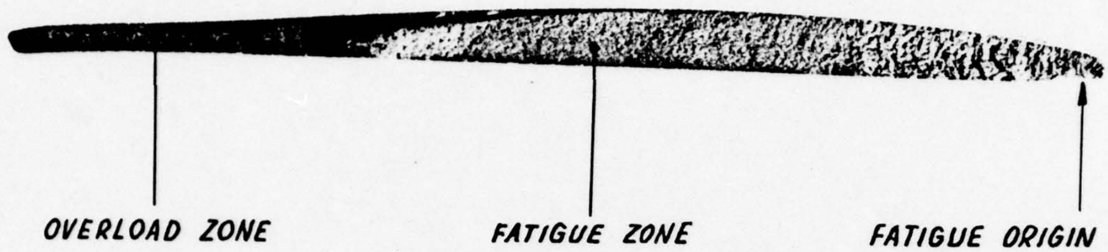


FIG. 9.3 PROPELLER TIP FAILURE SHOWING STONE DAMAGE IMPROPERLY DRESSED OUT
(Magn. x 0.8 approx.)

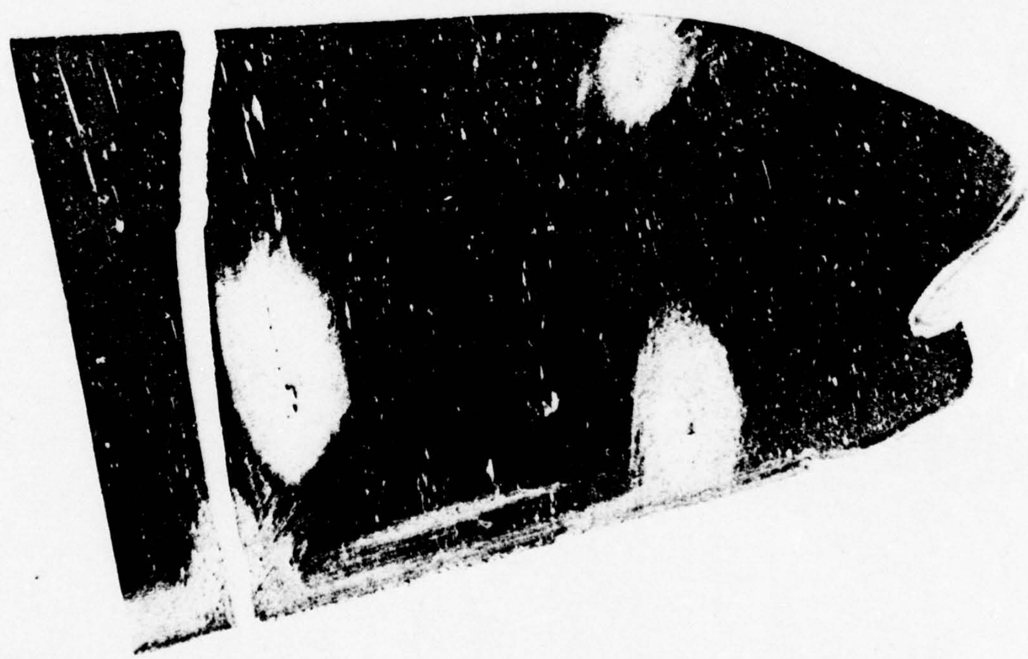


FIG. 9.4 PROPELLER TIP FAILURE – FRACTURE SURFACE
(Magn. x 1.5 approx.)

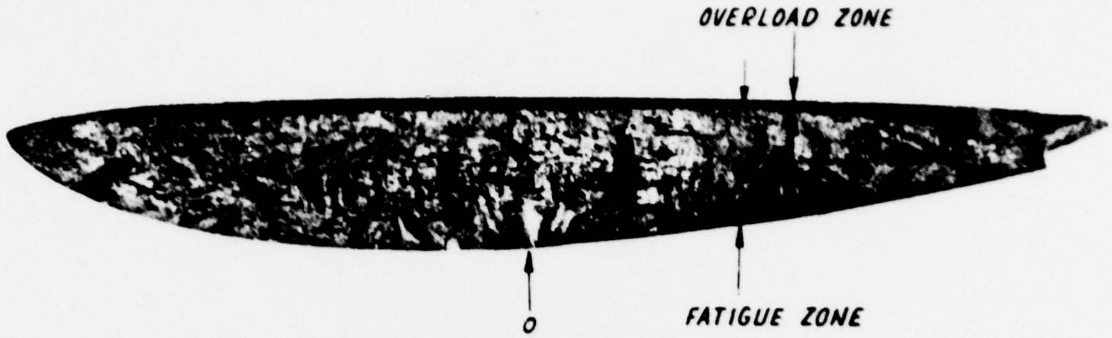


FIG. 9.5 PROPELLER FAILURE INITIATING AT CORROSION ATTACK
(Magn. x 1.1 approx.)

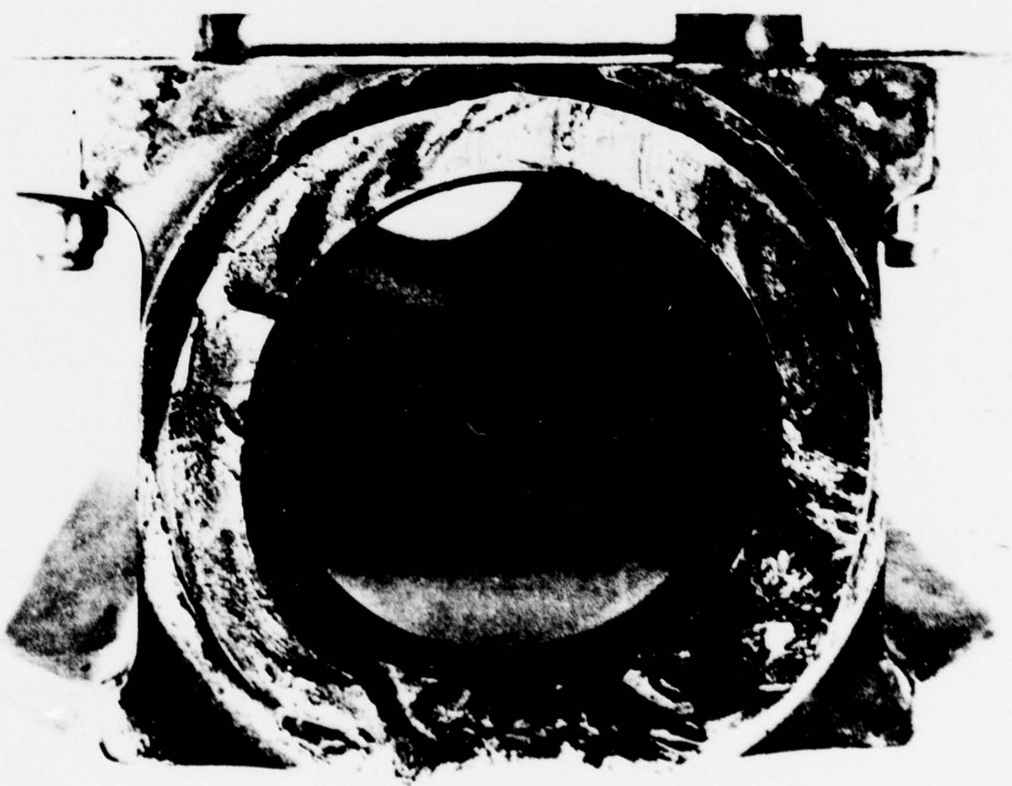


FIG. 9.6 FRACTURE SURFACE, HELICOPTER MAIN ROTOR YOKE.
(Magn. x 0.8 approx)

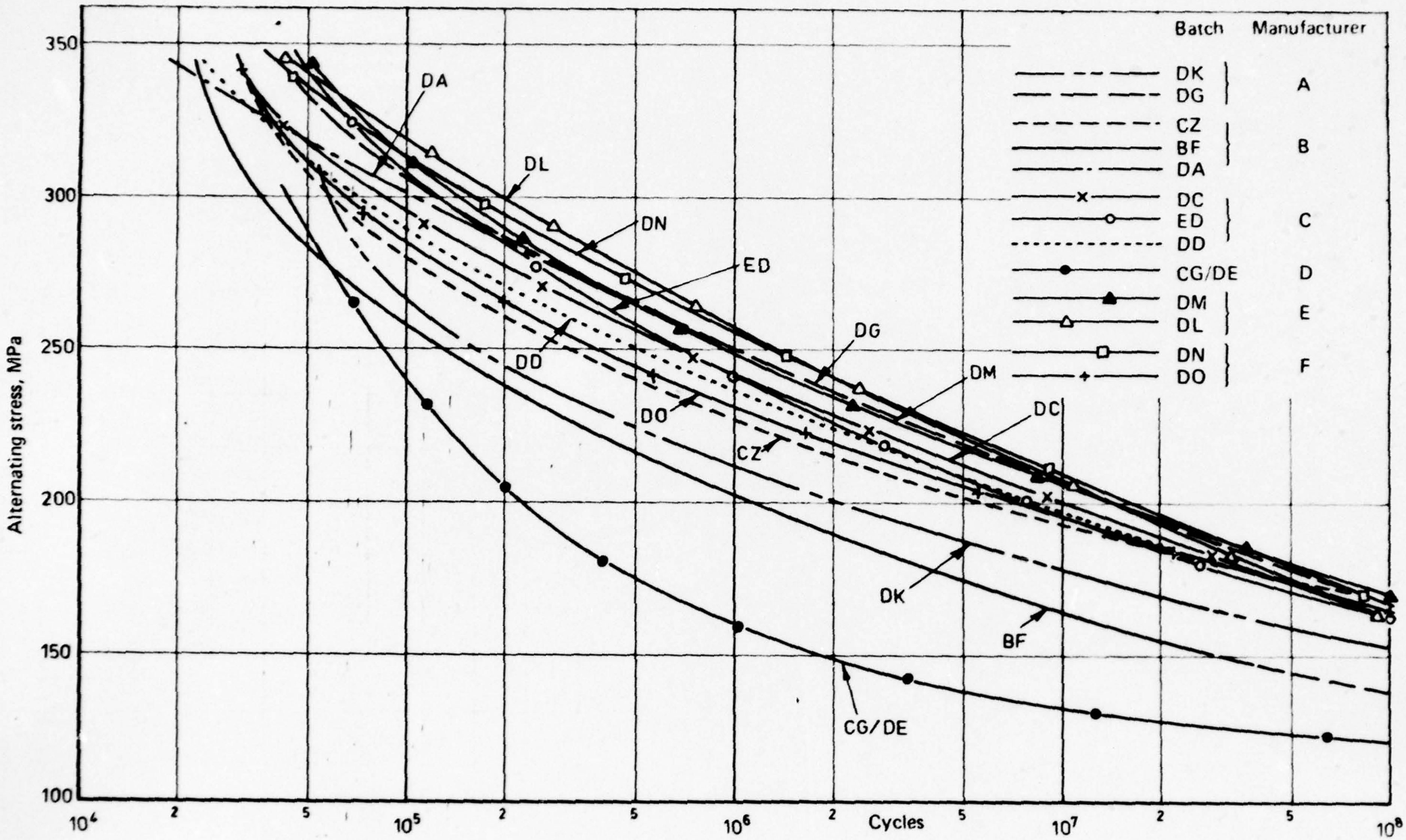


FIG. 9.7 COMPOSITE UNNOTCHED S/N CURVES FOR 13 BATCHES OF 2L.65 MATERIAL

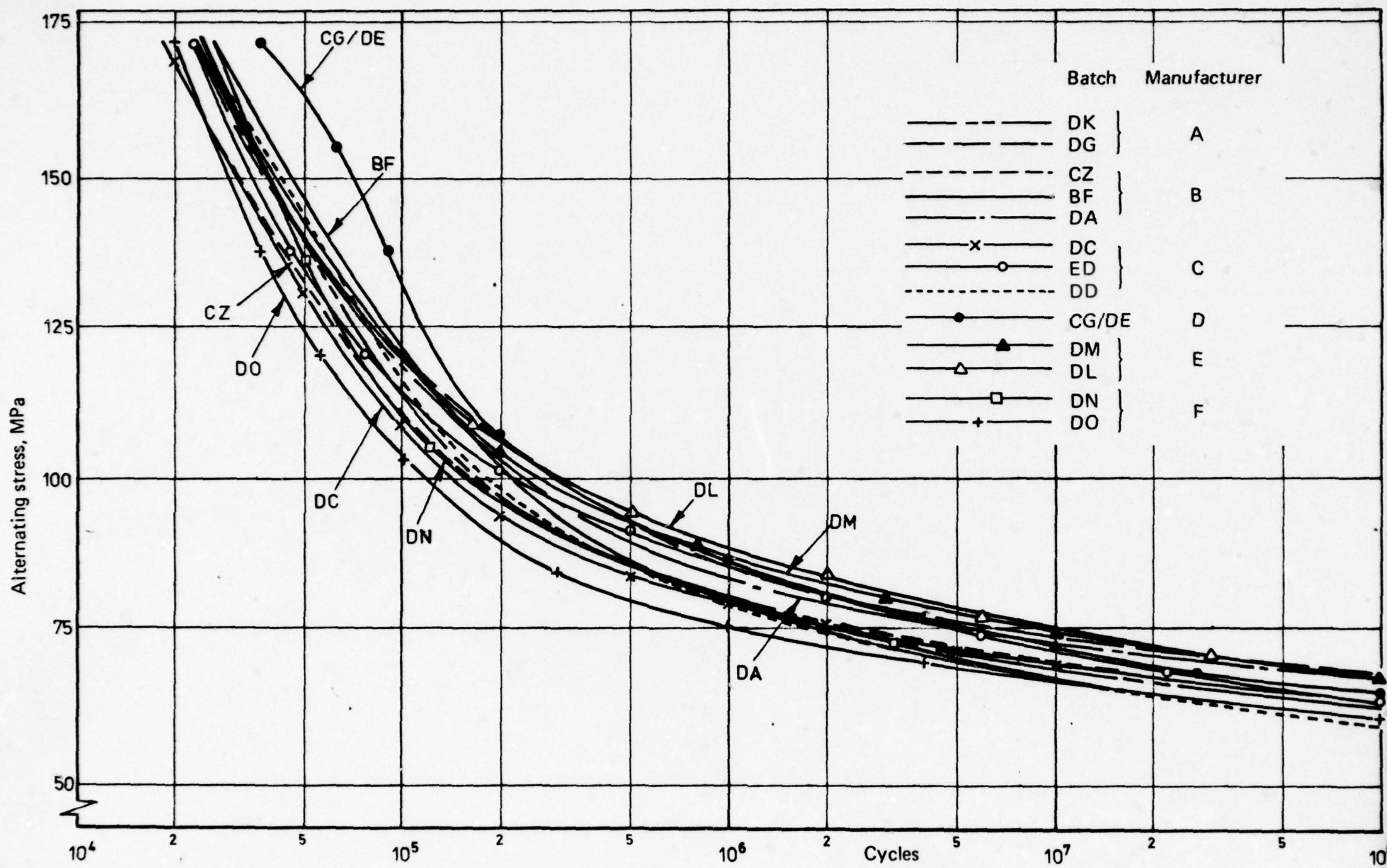


FIG. 9.8 COMPOSITE NOTCHED S/N CURVES FOR 13 BATCHES OF 2L.65 MATERIAL

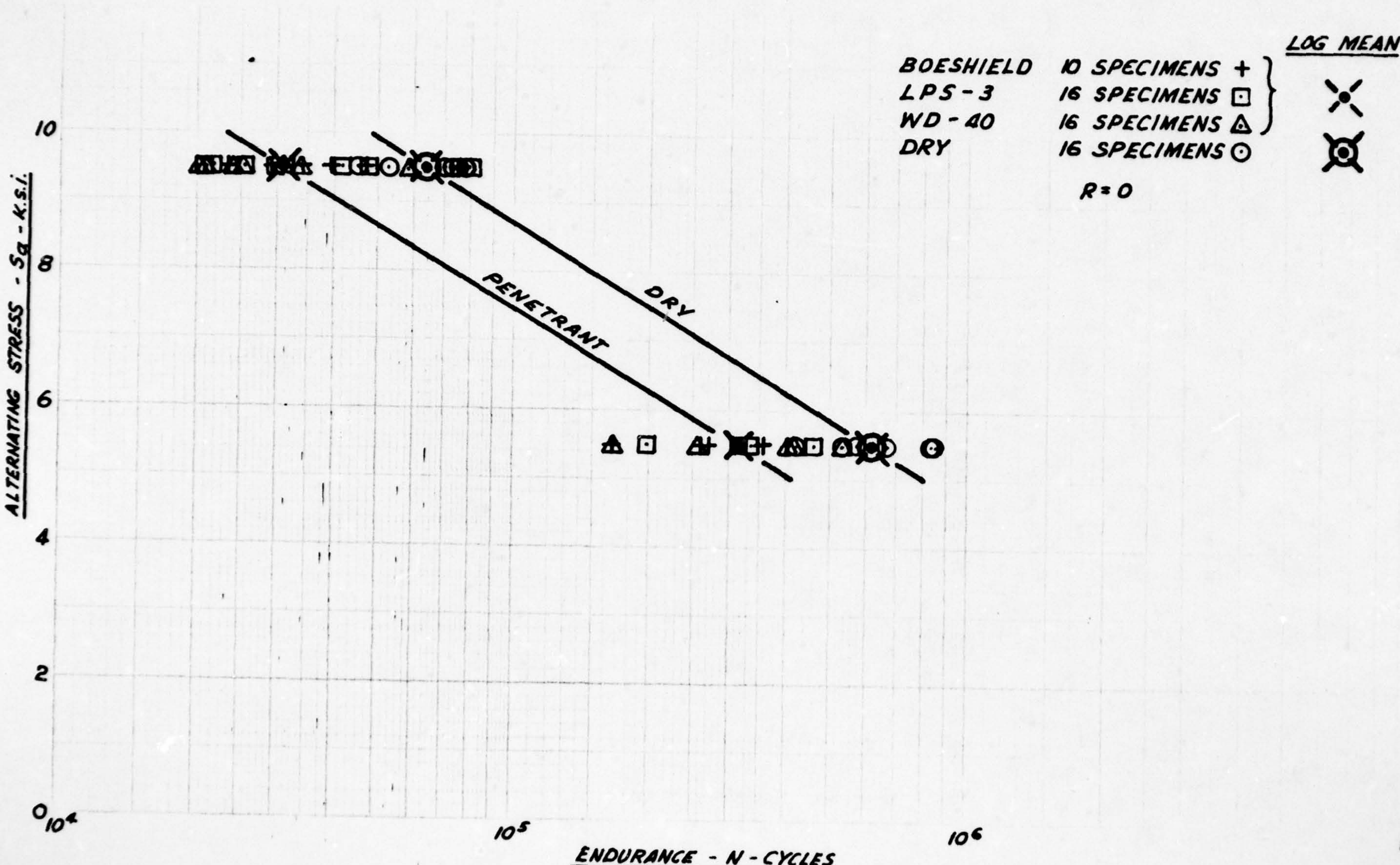
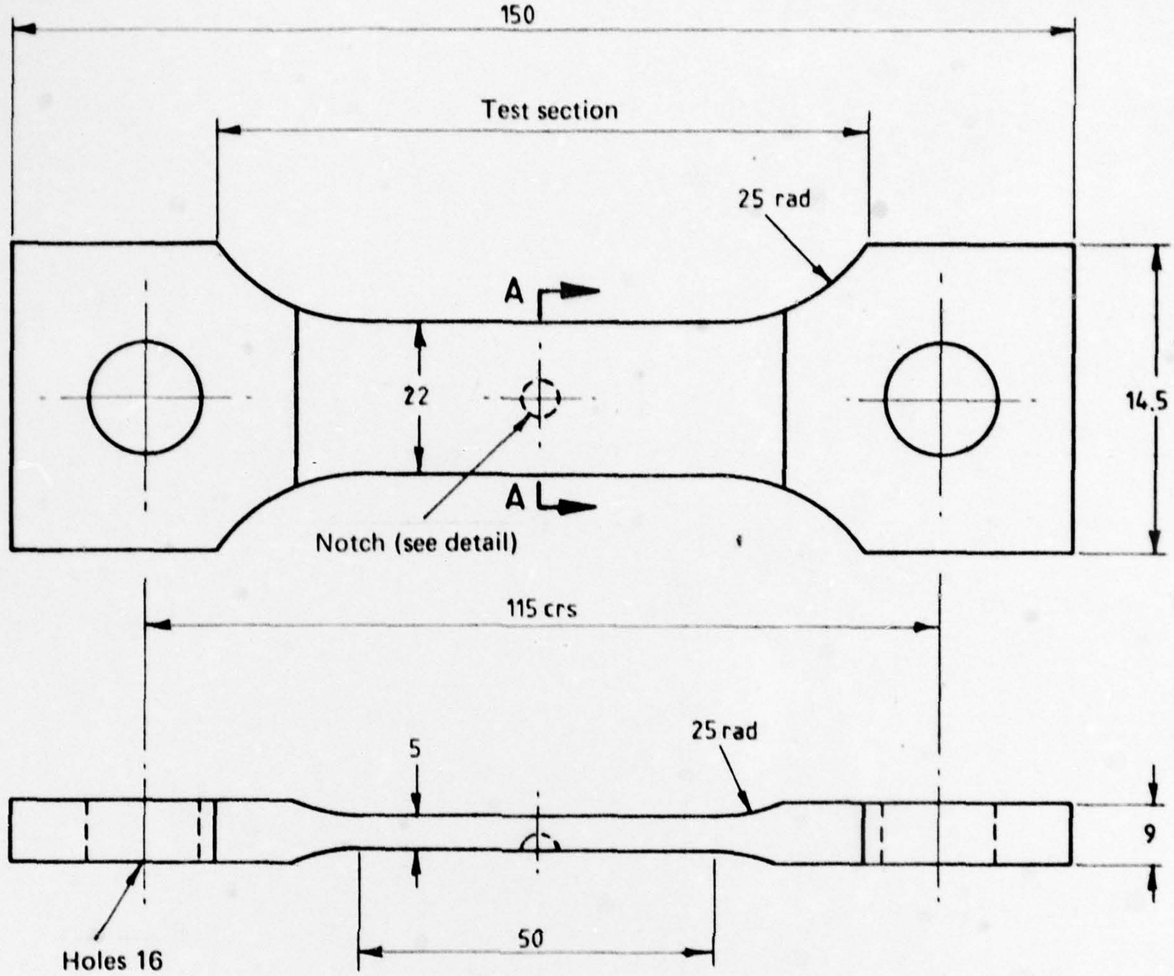
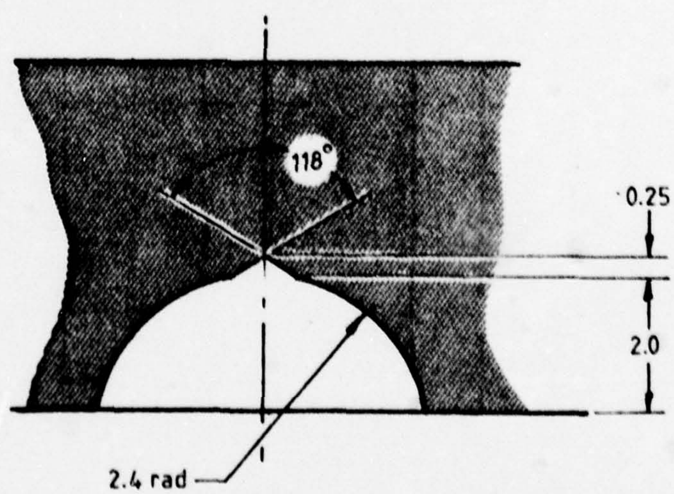


FIG. 9.9 FATIGUE TEST RESULTS – RIVETED LAP JOINT SPECIMENS





Section A-A
Details of notch
Scale 10:1

Dimensions in mm

FIG. 9.10 FATIGUE SPECIMEN

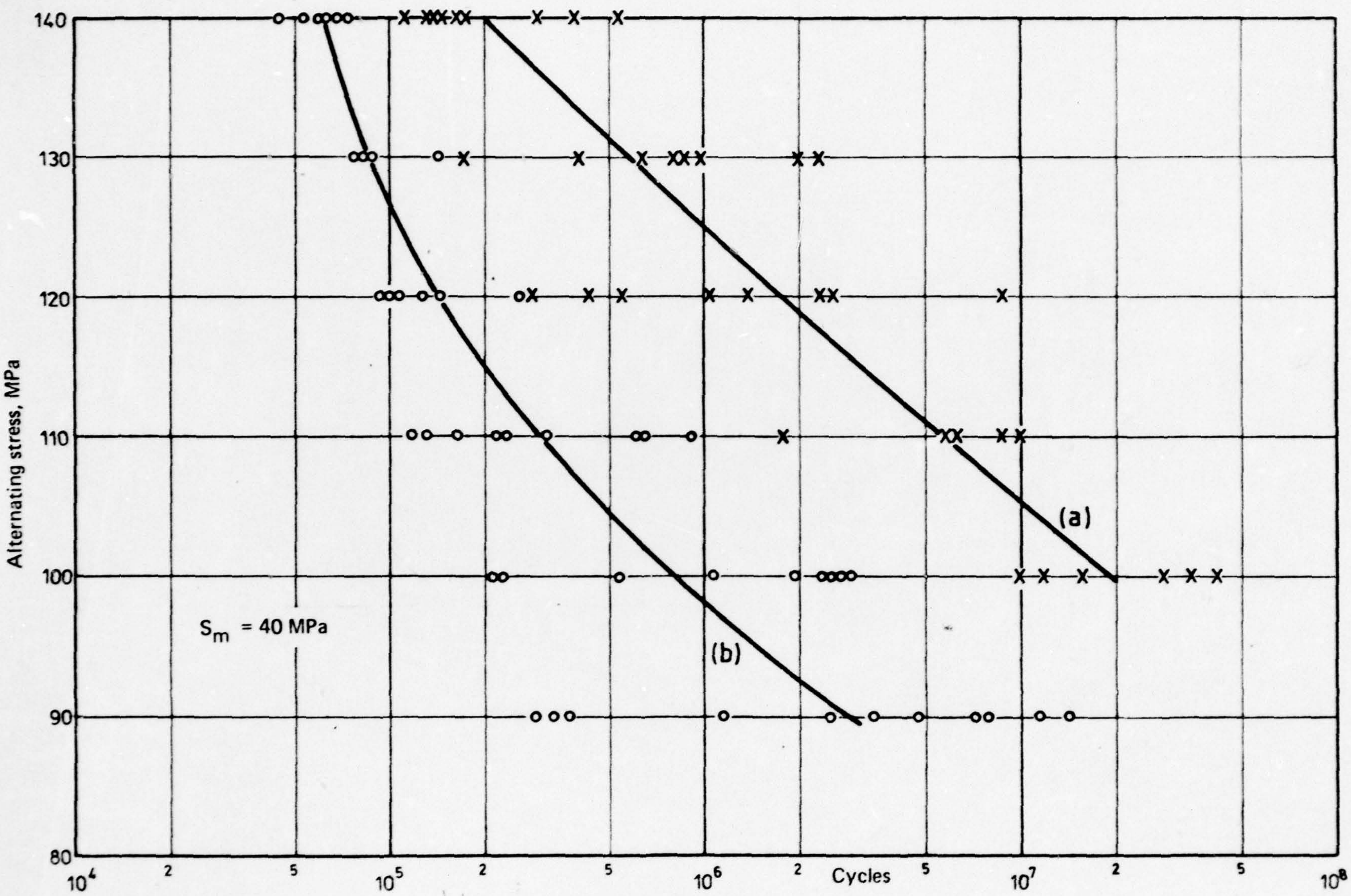


FIG. 9.11 FATIGUE DATA
(a) Spherical hole specimens (b) Drill point notch specimens

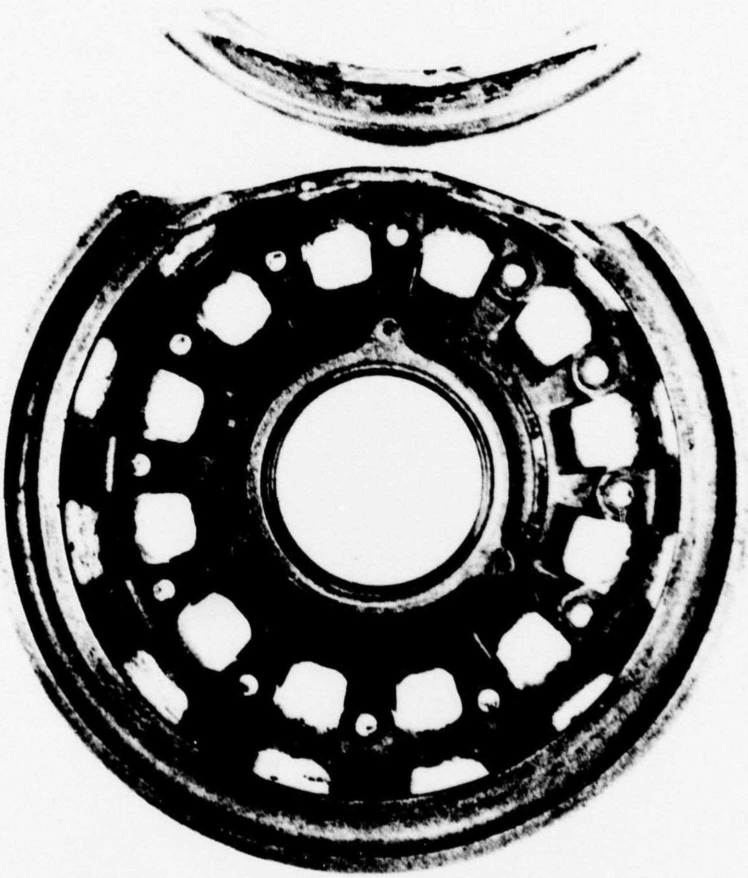


FIG. 9.12 GENERAL VIEW OF LANDING WHEEL FAILURE
(Magn. x 0.2 approx.)

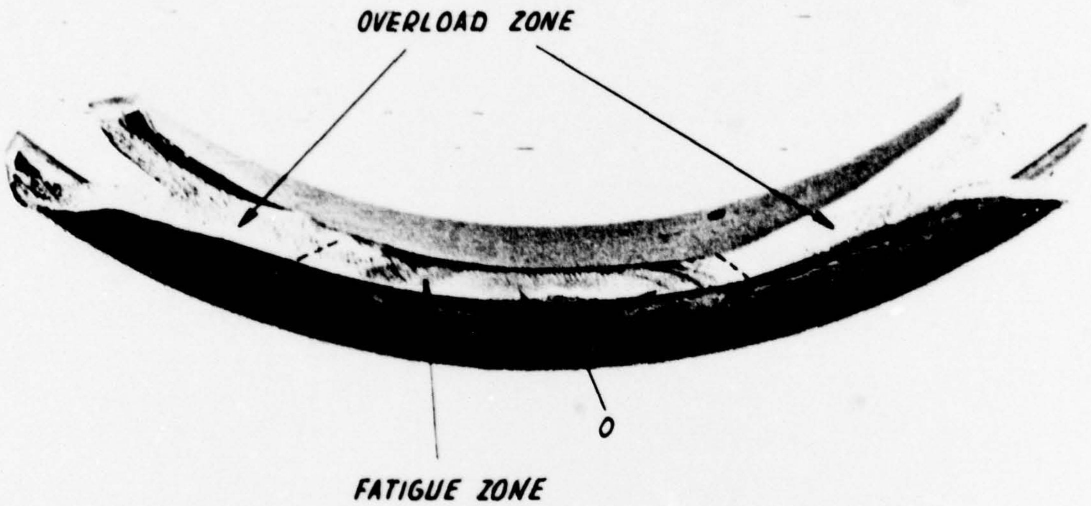


FIG. 9.13 TYPICAL LANDING WHEEL FRACTURE SURFACE
(Magn. x 0.45 approx.)

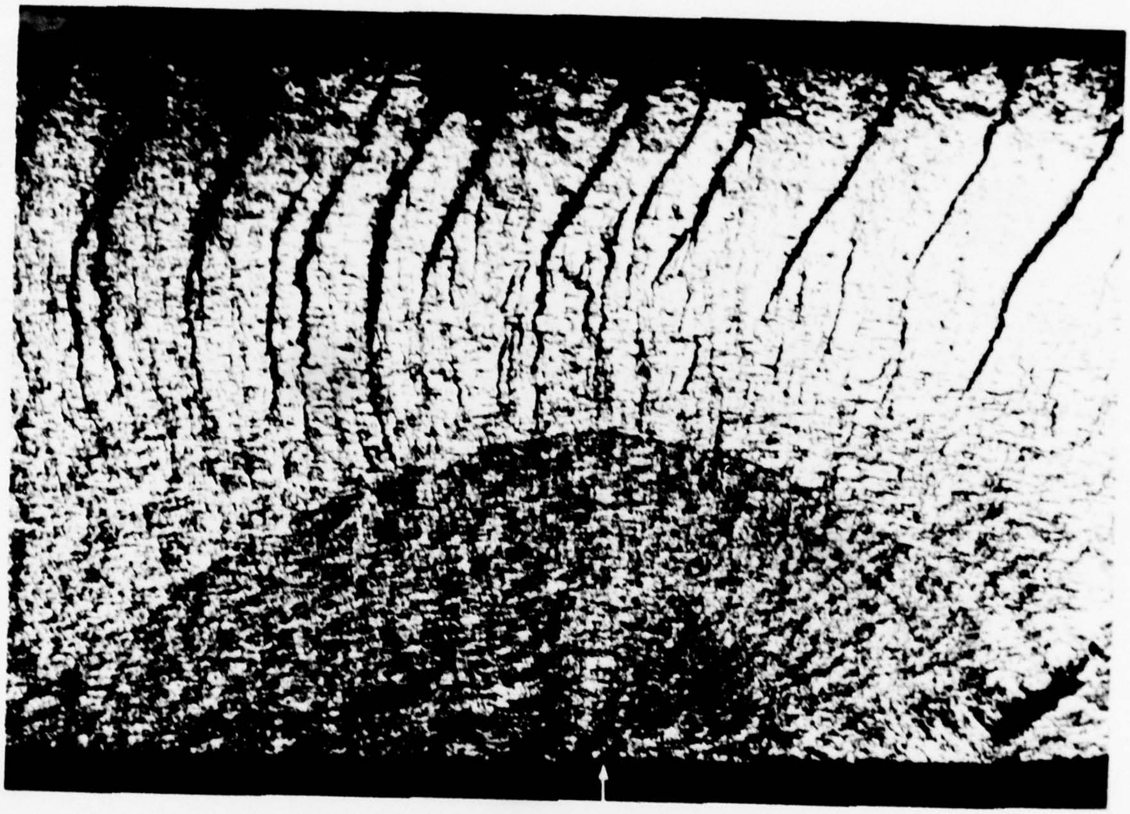


FIG. 9.14 SHOWS WHEEL FRACTURE INITIATION AT PITTING CORROSION
(Magn. x 9 approx.)

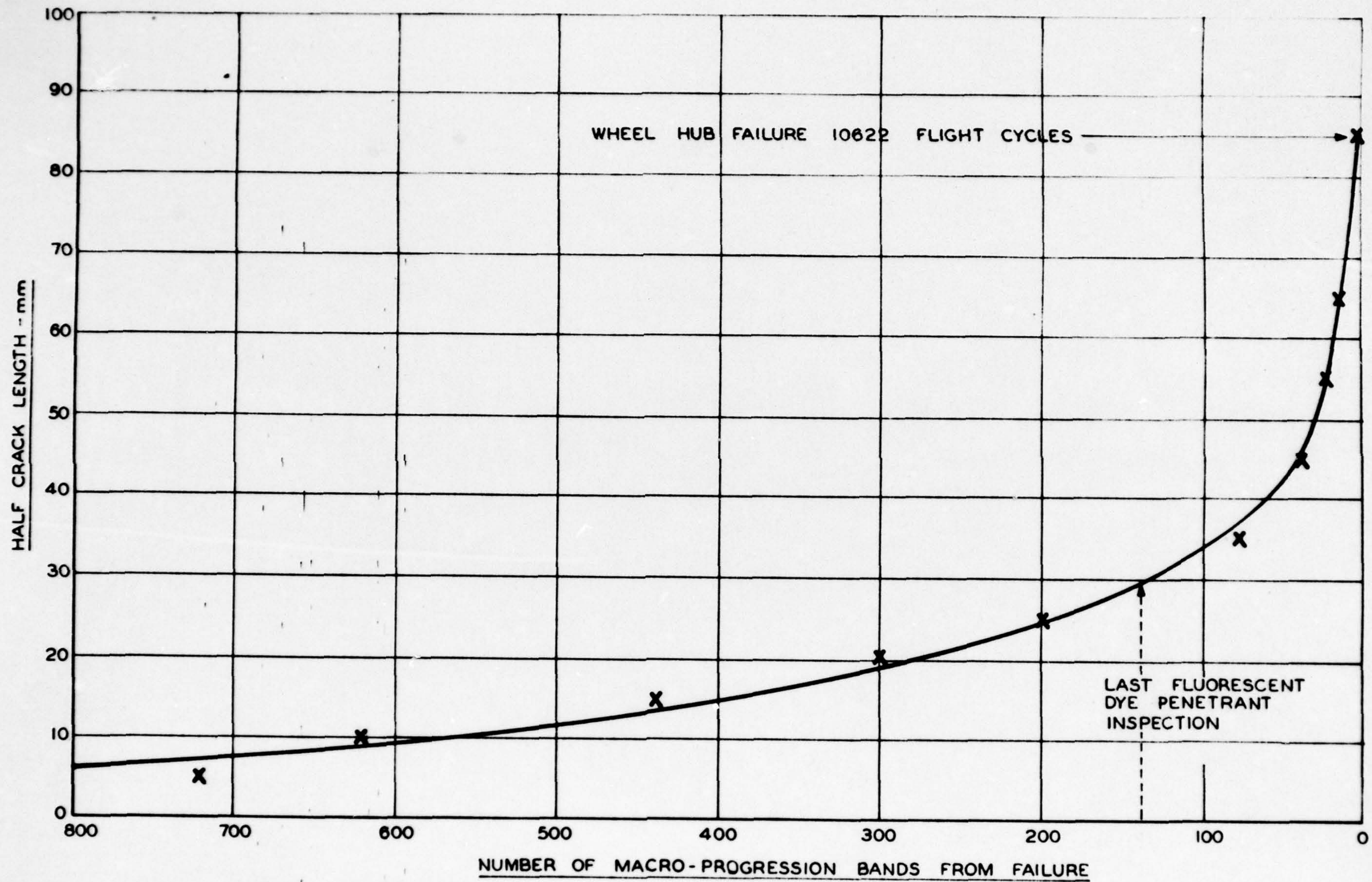


FIG. 9.15 FATIGUE CRACK GROWTH IN MAIN WHEEL OUTER HUB

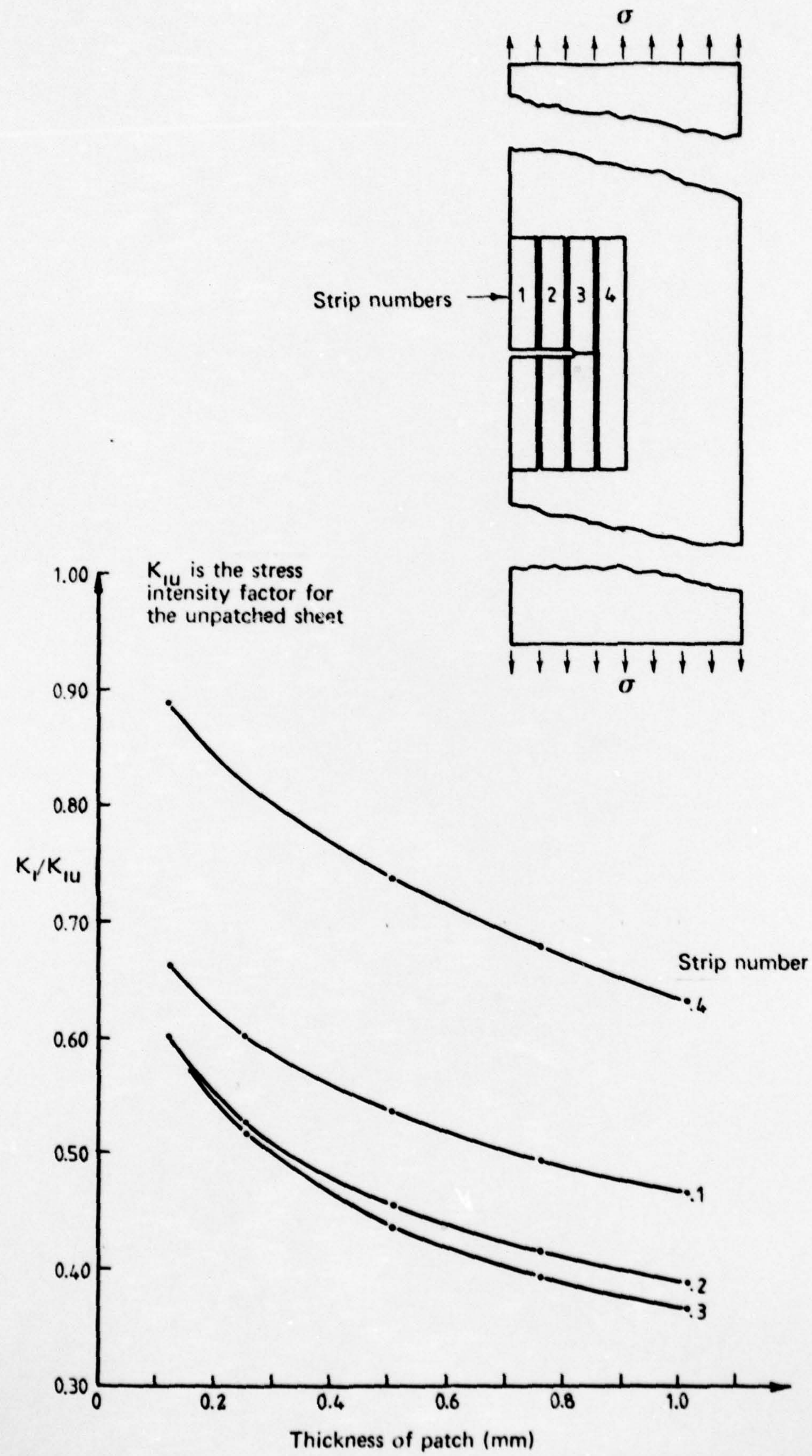


FIG. 9.16 EFFECT OF PATCH LOCATION ON THE STRESS INTENSITY FACTOR

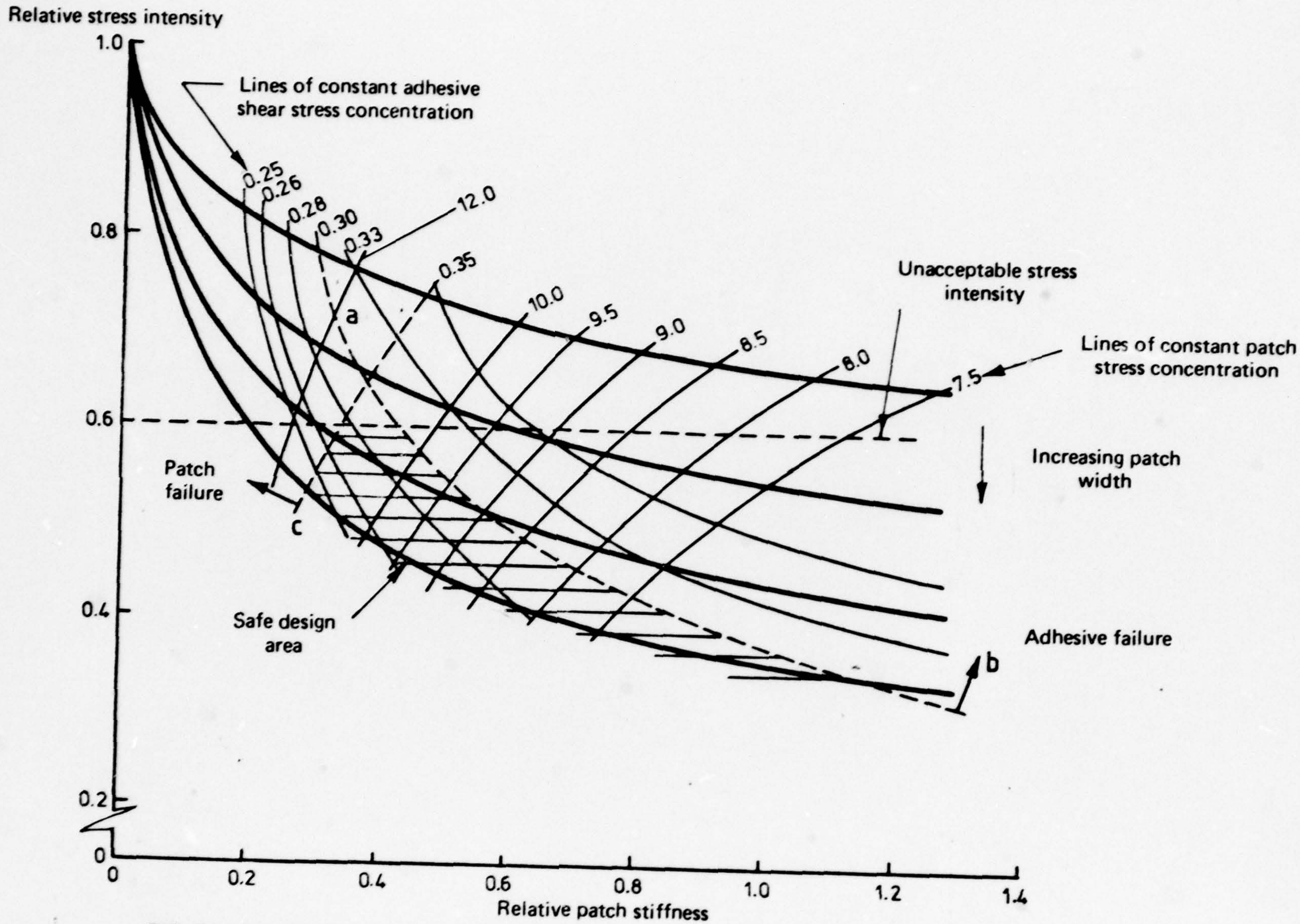


FIG. 9.17 SCHEMATIC EXAMPLE OF DESIGN CURVES FOR PATCHES BONDED AT THE CRACK TIPS OF A CENTRE CRACKED PANEL

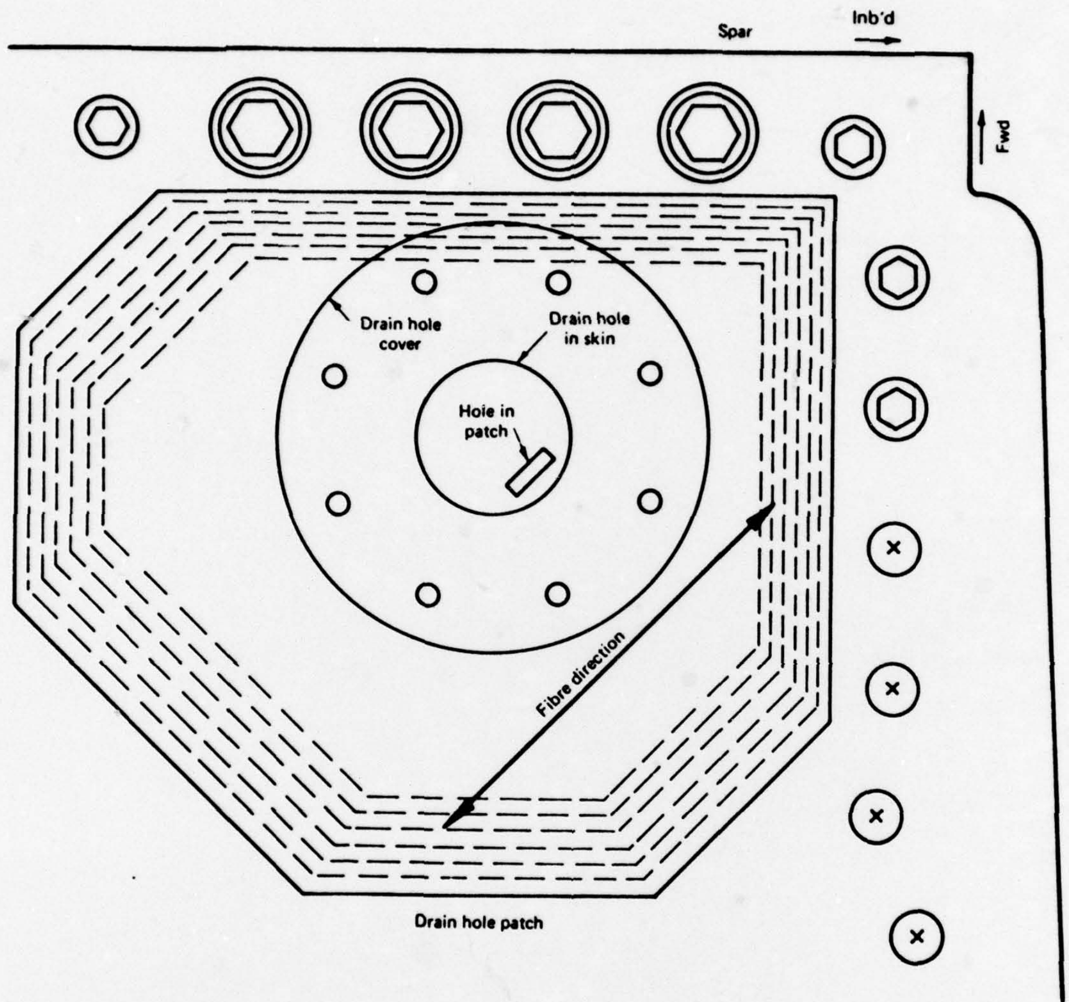


FIG. 9.18 GENERAL ARRANGEMENT OF BFRP PATCH FOR MIRAGE DRAIN HOLE REGION (¾ full size)

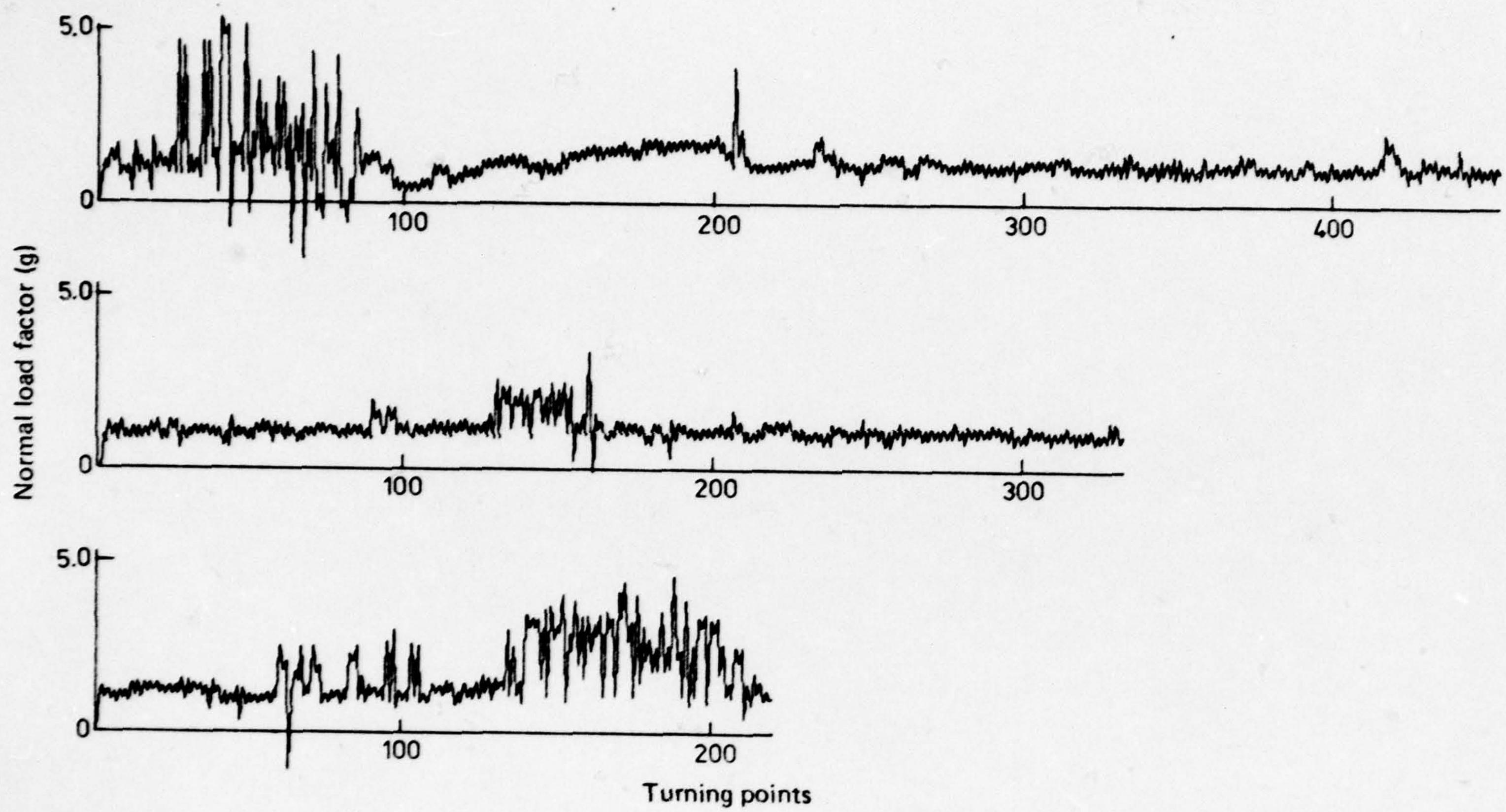


FIG. 9.19 EXAMPLES OF FLIGHTS FROM FLIGHT–BY–FLIGHT SEQUENCE

DOCUMENT CONTROL DATA SHEET

Security classification of this page: UNCLASSIFIED

1. DOCUMENT NUMBERS
a. AR Number:
AR-001-256610
b. Document Series and Number:
Structures Technical Memorandum 303
c. Report Number:
ARL-STRUC-TECH-MEMO-303

2. SECURITY CLASSIFICATION
a. Complete document:
UNCLASSIFIED
b. Title in isolation:
UNCLASSIFIED
c. Summary in isolation:
UNCLASSIFIED

3. TITLE:
A REVIEW OF AUSTRALIAN INVESTIGATIONS ON AERONAUTICAL FATIGUE DURING THE PERIOD APRIL 1977 TO MARCH 1979.

4. PERSONAL AUTHOR:

G.S. JOST

5. DOCUMENT DATE:
APRIL, 1979

6. TYPE OF REPORT AND PERIOD COVERED:

7. CORPORATE AUTHOR(S):
Aeronautical Research Laboratories

8. REFERENCE NUMBERS
a. Task:
RD73
b. Sponsoring Agency:
SUPPLY 50/2

9. COST CODE:
27003

10. IMPRINT:
Aeronautical Research Laboratories, Melbourne

11. COMPUTER PROGRAM(S)
(Title(s) and language(s)):

12. RELEASE LIMITATIONS (of the document)
Approved For Public Release.

12-0. OVERSEAS:	N.O.	1	P.R.	X	A	X	B	X	C	X	D	X	E	X

13. ANNOUNCEMENT LIMITATIONS (of the information on this page):

No Limitation

14. DESCRIPTORS:
Aircraft
Fatigue (Materials)
Structures
Reviews Australia

15. COSATI CODES:
0103
1113
2012
1407
0502

16. ABSTRACT:

This document was prepared for presentation to the 16th Conference of the International Committee on Aeronautical Fatigue scheduled to be held at Brussels, Belgium on May 14 and 15, 1979. A summary is presented of Australian aircraft fatigue research and associated activities. The major topics discussed include the fatigue of both civil and military aircraft structures, fatigue of materials and components and fatigue life monitoring and assessment.

DISTRIBUTION

	COPY NO.
AUSTRALIA	
Department of Defence	
Central Office	
Chief Defence Scientist	1
Deputy Chief Defence Scientist	2
Superintendent, Science and Technology Programs	3
Australian Defence Scientific and Technical Representative (UK)	4
Counsellor, Defence Science (USA)	5
Joint Intelligence Organisation	6
Defence Library	7
Assistant Secretary, D.I.S.B.	8-23
Aeronautical Research Laboratories	
Chief Superintendent	24
Superintendent, Structures Division	25
Divisional File, Structures	26
Author: G.S. Jost	27
Dr. A.O. Payne, Structures Division	28
Mr. J.Y. Mann, " "	29
Dr. D.G. Ford, " "	30
Mr. C.K. Rider, " "	31
Mr. C.A. Patching, " "	32
Mr. B.C. Hoskin, " "	33
Dr. J.M. Finney, " "	34
Mr. P.J. Howard, " "	35
Mr. P.H. Townshend, " "	36
Mr. B.E. Anderson, " "	37
Mr. J.M. Grandage, " "	38
Dr. A.O. Payne, (For TTCP Distribution)	39-43
Dr. G.S. Jost, (For ICAF Distribution)	44-84
Superintendent, Materials Division	85
Dr. A.A. Baker, " "	86
Dr. N.E. Ryan, " "	87
Dr. M.M. Hutchison, " "	88
Library	89
Materials Research Laboratories	
Library	90
Defence Research Centre Salisbury	
Library	91
Central Studies Establishment	
Information Centre	92

DISTRIBUTION (CONTD.)

	COPY NO.
Engineering Development Establishment	
Library	93
RAN Research Laboratory	
Library	94
Air Office	
Aircraft Research & Development Unit, Scientific Flight Group	95
Air Force Scientific Adviser	96
Engineering (CAFTS) Library	97-100
D.Air Eng. - AF	101-102
HQ Support Command (SENGSO)	103
Army Office	
Army Scientific Adviser	104
Royal Military College Library	105
Navy Office	
Naval Scientific Adviser	106
Directorate of Naval Aircraft Engineering, (Mr. W. Wodrow)	107
Department of Productivity	
Government Aircraft Factories	
Manager/Library	108-109
Victorian Region	
Library	110
Department of Transport	
Secretary/Library	111
Airworthiness Group, Mr. K. O'Brien	112
Airworthiness Group, Mr. R.B. Douglas	113
Airworthiness Group, Mr. G.P. Esson	114
Australasian ICAF Distribution	
Victoria	
Dr. R.H. Keays, M.S. 10, Commonwealth Aircraft Corporation	115
Mr. C. Torkington, Principal Airworthiness Engineer (Fatigue), Airworthiness Branch, Department of Transport	116
Mr. T.F. Harvey, Technical Librarian, Trans-Australia Airlines	117
Mr. G.D. Bennett, Chief Engineer, Government Aircraft Factories	118
Mr. J.H. Bibo, Technical Manager, Ansett Airlines of Australia	119
Mr. H.K. Millicer, Department of Aeronautics R.M.I.T.	120

DISTRIBUTION (CONTD)

	COPY NO.
Australasian ICAF Distribution (contd.)	
New South Wales	
Mr. J.M. Whitney, Chief Aeronautical Engineer, Hawker de Havilland Australia, Pty. Ltd., Bankstown	121
Mr. R.H. Stevens, Chief Structures Engineer, Qantas, Qantas Jet Base, Qantas, Mascot	122
New Zealand	
Mr. J.E.C. Shearer, Departmental Librarian, Department of Mech. Eng., University of Canterbury	123
Mr. D.J. O'Brien, Chemistry Division, Department of Scientific and Industrial Research, Petone	124
Mr. E.T. Labett, Civil Aviation Division, Department of Transport	125
Within Australia	
Library, Qantas	126
Library, Ansett Airlines of Australia	127
Library, Hawker de Havilland Australia Pty. Ltd., Bankstown	128
Library, Hawker de Havilland Australia Pty. Ltd., Lidcombe	129
SPARES	130-139